WOODS HOLE OCEANOGRAPHIC INSTITUTION

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Calibrating PBY-6A Airplane For Meteorological Purposes

By

Given A. Brewer
Consultant

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Director

PREFACE

With the loan, through the courtesy of the Office of Naval Research, of a PBY-6A aircraft to this project for the measurement and investigation of atmospheric turbulence and convection, numerous problems have arisen concerning the use of this airplane as a meteorological tool and concerning its instrumentation. These problems center around the detection of atmospheric motions, such as drafts and gusts, from an extended object which is itself in motion through the air. Much information concerning the aerodynamics and structural characteristics of the aircraft itself has been needed in order to interpret reliably the readings of the meteorological instruments that are mounted on and within it.

For these purposes a consulting engineer, Given A. Brewer, has been retained by the project. The present report constitutes some further results of the very essential studies of the aircraft made by him. It is identical with Given Brewer Report 075.

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ABSTRACT

As part of a continuing effort to reconstruct gust profiles during cloud traverses, an experimental program was carried out calibrating sensing instruments within a PBY-6A. Using multi-channel oscillographic equipment, four transducing instruments monitoring the aircraft's behavior were simultaneously observed. Vector airspeed, Cg acceleration, pitch attitude, and altitude were continuously recorded oscillographically. The various transducing instruments were calibrated in the laboratory at the Woods Hole Oceanographic Institution and then transferred to a Navy PBY-6A for flight testing.

The aircraft was subjected to a number of maneuvers in smooth air so that the performance of the instruments might be checked under known conditions. A number of steep pull-ups were executed and the oscillographic records for these maneuvers have been analyzed. It has been determined that for rates of vertical acceleration considerably greater than encountered during meteorological research, the velocities predicted from acceleration measurements agree with those deduced from altitude changes.

Again, within the range of velocity profiles sought, departures between the dynamic and static lift coefficient, due to circulation lag, have been found negligible. For the practical purposes of meteorological research, the PBY-6A has been found to behave as a rigid body with no discernable errors due to wing whip. It was found that the placement of accelerometers within the fuselage of the PBY is not critical in the determination of vertical acceleration.

INTRODUCTION

During the past 21 years the gust structure of the atmosphere has been investigated by means of suitably instrumented aircraft. Early experiments[1] employed mirrors fastened to wings of airplanes such that deflections in flight would cause light spot movements on photosensitive paper and thus permit analytical reconstruction of the air loads encountered, during flight through turbulence. During the intervening years instrumentation has improved and research[2,3,4,5,6] has increased understanding of the relationship between atmospheric turbulence and resultant motions of the sounding aircraft.

This present study represents a continuing effort to reconstruct wing air loads from observations of speed, altitude, acceleration and pitch angle, oscillographically recorded within the aircraft. The problems of properly locating accelerometers to predict wind loads in a PBY-6A have been investigated in an earlier work[7]. Placement of pressure sensitive instruments to avoid interference effects from the aircraft is difficult. Departures of the lift coefficient from its steady state value are to be expected under dynamic flight conditions[8], yet certain knowledge of the actual lift coefficient at any instant is required to predict gust velocities from aircraft accelerations. Acceleration errors due to wing whip may be appreciable and are functions of gust frequency and wing rigidity[9].

This report describes the experimental procedures designed to prove validity of the instrumental techniques and

installations within the PBY-6A NAVY 46683. This airplane is currently being used for meteorological soundings by the Woods Hole Oceanographic Institution under contract from the Office of Naval Research.

DETERMINING LIFT AND DRAG COEFFICIENTS

The PBY-6A has a small spirit level in the engineer's compartment. This level is so oriented that the angle between the wing chord plane and the ground is read directly. The aircraft was flown at constant speed at various attitudes with observations made of manifold pressure, engine rpm, altitude, air speed, outside air temperature and angle of attack. This procedure was repeated a number of times with a summary of the data and the steady state lift coefficient carried in the Table I. From information supplied by the manufacturers, calculations of engine hp and propeller efficiency were made permitting determination of the over-all airplane drag coefficient, Table II. The theoretical plot of lift coefficient vs angle of attack for the NACA 21 section has been obtained from Ref. 10, corrected for aspect ratio and plotted on the Figure 1. A comparison between the theoretical and measured values of steady flight lift coefficient shows close correlation.

To determine the dynamic lift coefficient, the airplane was subjected to a number of abrupt maneuvers with changes in air speed, acceleration, altitude and pitch angle observed oscillographically. The abrupt pull-ups were executed in still air so that the instrument recordings could be checked against known conditions.

TABLE I

CALCULATING AIRPLANE LIFT COEFFICIENT, C_L - PBY-6A

$$C_L = W/S \cdot q$$

DATE P_o Hg"	V_i mph	H_p ft	t °C	W lbs	Cg inches	α_c deg	q lbs/ft²	C_L
	80.7	5240	9.0	29641	256.5	12.8	16.66	1.271
9-8-52	92.2	5200	9.0	29641	256.5	9.2	21.75	.973
30.52"	103.7	5200	9.0	29641	256.5	8.3	27.50	.770
	115.2	5220	9.0	29641	256.5	5.8	34.00	.622
	126.9	2100	11.0	29641	256.5	4.5	41.30	.513
	80.7	1800	0.0	28986	253.0	15.0	16.66	1.240
10-20-52	92.2	1850	0.0	28986	253.0	10.5	21.75	.951
29.92"	103.7	1800	0.0	28986	253.0	8.0	27.50	.752
	115.2	1700	0.0	28986	253.0	5.5	34.00	.608
	126.8	1800	0.0	28986	253.0	4.5	41.30	.501
	138.2	1850	0.0	28986	253.0	3.5	49.00	.422
	101.3	7000	14.5	29353	250.0	8.0	26.30	.792
3-18-53	115.2	7000	14.5	29353	250.0	6.0	34.00	.613
29.98"	126.8	7000	14.5	29353	250.0	5.0	41.30	.505
	138.1	7000	14.5	29353	250.0	3.0	49.00	.425
	103.7	6100	10.0	29923	249.3	-	27.50	.779
3-21-53	115.2	6100	10.0	29923	249.3	-	34.00	.630
30.05"	126.8	6100	10.0	29923	249.3	4.5	41.30	.518
	126.8	6100	10.0	29923	249.3	-	41.30	.518
	126.9	6700	17.0	29908	250.5	4.0	41.30	.518
4-2-53	103.8	6700	17.0	29908	250.5	7.5	27.50	.779
29.94"	115.2	6700	17.0	29908	250.5	6.0	34.00	.630
	92.3	6700	17.0	29908	250.5	9.5	21.75	.986
	92.3	10500	10.0	29231	250.0	11.0	21.75	.961
4-5-53	103.7	10500	10.0	29231	250.0	8.0	27.50	.760
30.07"	115.2	10500	10.0	29231	250.0	5.5	34.00	.615
	126.8	10500	10.0	29231	250.0	4.0	41.30	.506

NOTES:
P_o = sea level pressure (Hg")
V_i = indicated air speed (mph)
H_p = pressure altitude (ft)
t = air temperature (°C); t_o = surface temperature = 15°C
W = gross weight of airplane (lbs)
S = wing area (ft²)
Cg = airplane center of gravity (inches aft of nose)
α_c = chord plane angle of attack (deg)
q = $0.00256\ V_i^2$ (lbs/ft²)

TABLE II

CALCULATING OVER-ALL AIRPLANE DRAG COEFFICIENT, C_d - PBY-6A

$$C_d = T/S \cdot q$$

Date		10-20-52						3-21-53
P_o	(Hg")	29.92"						30.05"
V_i	(mph)	80.7	92.2	103.7	115.2	126.8	138.2	126.8
V_a	(mph)	81.8	93.5	105.0	117.0	128.3	140.1	140.8
MAN	(Hg")	25.0	25.0	25.0	28.0	32.0	34.0	30.0
Eng.	(rpm)	2000	2000	2000	2000	2100	2300	2000
H_p	(ft)	1800	1850	1800	1700	1800	1850	6100
t	(°C)	0	0	0	0	0	0	10
t_{std}	(°C)	11.0	11.0	11.0	11.0	10.8	10.4	11.0
W	(lbs)	28986	28986	28986	28986	28986	28986	29923
α_c	(deg)	15.0	10.5	8.0	5.5	4.5	3.5	4.5
N	(rpm)	1125	1125	1125	1125	1180	1292	1125
q	(lbs/ft^2)	16.66	21.75	27.50	34.00	41.30	49.00	41.30
BHP		912	912	912	1042	1323	1510	1210
C_p		.131	.131	.131	.149	.164	.142	.207
J		.533	.600	.676	.762	.797	.795	.918
ρ/ρ_o		.982	.982	.982	.982	.982	.982	.821
η		.642	.670	.694	.706	.700	.728	.675
T	(lbs)	2685	2450	2260	2360	2710	2940	2170
C_d		.1152	.080	.059	.050	.0465	.0425	.0375

NOTES V_a = true air speed (mph)
MAN = manifold pressure (Hg")
t_{std} = temperature of standard atmosphere
N = propeller (rpm)
BHP = brake horsepower

$$C_p = \frac{2.0 \text{x} BHP \text{x} 10^5}{N^3 \ \rho/\rho_o} ; \quad J = \frac{7.33 \ V_a}{N} ; \quad \rho/\rho_o = \frac{Pt_o}{P_o t}$$

η = propeller efficiency
T = thrust = ηx375 BHP/V_a
PWA R-1830-92 engine gear reduction ratio 9:16 = .5625
Power ref. Fig. 71 Pilot's Handbook AN-01-5MC-1. PWA letter N. Larson to G. Brewer 5-28-53
Propeller efficiency curve P-10607. Hamilton Standard Propeller Blade 6353A-12 U.A.C. letter G. Rosen to G. Brewer 4-28-53

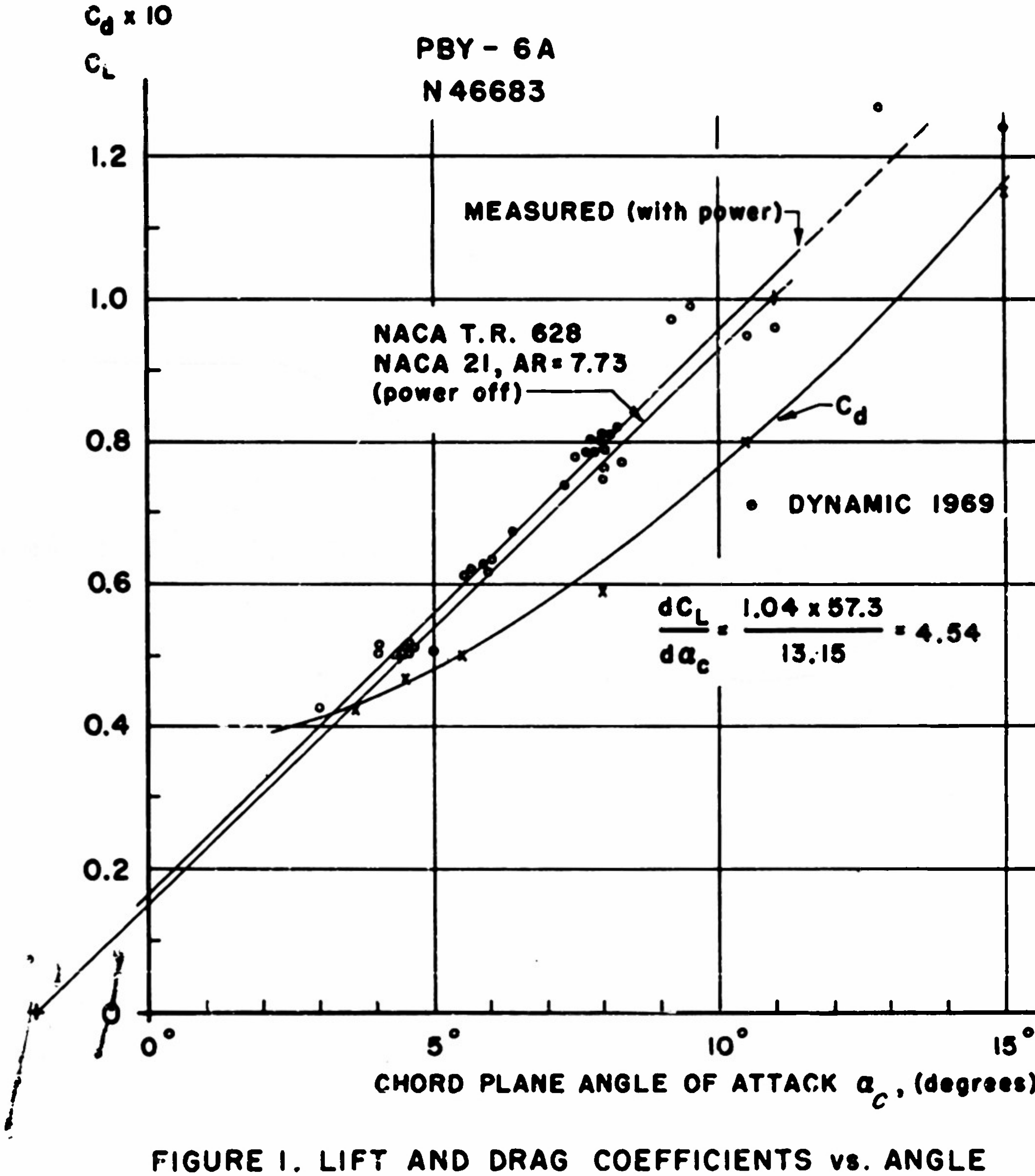

FIGURE 1. LIFT AND DRAG COEFFICIENTS vs. ANGLE OF ATTACK.

By integrating the accelerometer traces and by differentiating the altimeter trace, the airplane's vertical velocity vs time has been determined. From the oscillograph records of pull-ups nos. 1965 and 1969, the dynamic value of lift coefficient has been calculated: see Tables III and IV and Figure 1. For the run 1969 where a vertical airplane velocity of 32 ft/sec was achieved in 7 seconds, the dynamic value of lift coefficient corresponds at every point with the steady state value. For the more abrupt pull-up, run no. 1965, correlation between the unsteady and steady lift coefficients is not close. In run 1965, a vertical speed of 35 ft/sec was reached in 4 seconds. It is believed that the discrepancies between C_{LD} and C_L are due to lack of precision in the measurement of pitch angle at this high rate of pitch. As the meteorological gust spectrum being investigated envisions the sharpest gust rising at about 1½ ft/sec/sec in clear air turbulence and 3 ft/sec/sec in clouds, errors due to circulation lag may be neglegtected.

Referring to the Figures 2 and 3 it can be seen that although pitch angle, Υ, changes 12° to 14° the actual angle of attack, α_c, increases only 2° to 3° and holds constant during most of the maneuvers. As a consequence dynamic effects due to change in circulation must be confined to the first second or two at these maneuvers.

TABLE III

CALCULATING DYNAMIC VALUE OF LIFT COEFFICIENT, C_{LD} - RUN 1965

$$C_{LD} = (1 + \Delta N)W/S \cdot q$$

TIME	(sec)	0	.5	1.0	2.0	3.0	4.0	5.0	6.0
PITCH									
DEF	(in)	0	0.4	0.85	2.03	2.73	3.03	3.47	3.46
γ	(deg)	0	1.72	3.66	8.73	11.73	13.02	14.91	14.89
AIRSPEED									
DEF	(in)	0	0	0	-.18	-.61	-1.18	-1.70	-2.28
ΔTAS	(ft/sec)	0	0	0	-1.73	-5.86	-11.32	-16.31	-21.87
TAS	(ft/sec)	179.00	179.00	179.00	177.27	173.14	167.68	162.69	157.13
U_g	(ft/sec)	0	1.20	4.32	18.02	29.82	34.64	38.58	38.58
U_g/TAS	(deg)	0	.384	1.382	5.83	9.88	11.87	13.61	14.06
$\alpha_c = \gamma - U_g/TAS + 6°$		5.00°	7.34°	6.29°	8.80°	8.00°	7.60°	7.20°	7.02°
ACCELEROMETER									
DEF	(in)	0	0.64	1.70	2.37	1.22	0.73	0.33	-0.28
ΔN		0	.122	.325	.453	.233	.140	.063	-.054
IAS	(ft/sec)	169.10	169.10	169.10	167.46	163.56	158.39	153.68	148.40
q	(lbs/ft^2)	33.95	33.95	33.95	33.25	31.72	29.78	27.93	26.12
(1 + ΔN)W		30433	34200	40300	44300	37600	34700	32400	28800
C_{LD}		.641	.720	.849	.950	.850	.834	.829	.788
C_L		.641	.748	.827	.860	.795	.770	.738	.700

NOTES:

1. Test 3-13-53 3000 ft. 100 Knots entry speed
2. G. W. = 30433 lbs. = W
3. DEF (in) = Oscillograph screen deflection
   a. γ = Pitch (deg) = 4.3 °/in
   b. V_1 = 9.08 ft/sec/in
   c. Acceleration = 6.15 ft/sec^2/in
   d. Altimeter = 77.3 ft/in
4. TAS = 1.058 x IAS = 179 ft/sec
5. U_g = airplane vertical velocity ft/sec

TABLE IV

CALCULATING DYNAMIC VALUE OF LIFT COEFFICIENT. C_{LD} - RUN 1969

$$C_{LD} = (1 + \Delta N)W/S \cdot q$$

TIME (sec)	0.5	1.0	1.5	2.0	2.5	3.0	4.0	5.0	6.0	7.0
PITCH										
DEF (in)	.000	.145	.420	.740	1.130	1.400	1.910	2.450	2.800	3.010
γ (deg)	.000	.623	1.806	3.180	4.860	6.020	8.210	10.520	12.050	12.950
AIRSPEED										
DEF (in)	.00	.00	.00	.00	-.05	-.15	-.46	-.84	-1.28	-1.78
ΔTAS (ft/sec)	.000	.000	.000	.000	-.446	-1.340	-4.110	-7.500	-11.420	-15.900
TAS (ft/sec)	179.00	179.00	179.00	179.00	178.55	177.66	174.89	171.50	167.58	163.10
U_g (ft/sec)	.00	.25	1.65	4.70	8.76	12.57	19.68	25.26	29.57	31.85
U_g/TAS (deg)	.0000	.0802	.5280	1.5080	2.8100	4.0600	6.4500	8.4500	10.1000	11.0500
α_c	6.000	6.540	7.278	7.670	8.050	7.960	7.760	8.070	7.950	7.900
ACCELEROMETER										
DEF (in)	-.04	.29	.83	1.20	1.33	1.29	1.03	.85	.57	.08
ΔN	-.00765	.05550	.15880	.23000	.25400	.24700	.19700	.16260	.10900	.01530
IAS (ft/sec)	169.0	169.0	169.0	169.0	168.8	168.0	165.1	162.1	158.3	154.2
IAS (mph)	115.2	115.2	115.2	115.2	114.8	114.3	112.4	110.5	107.9	105.0
q (lbs/ft^2)	34.0	34.0	34.0	34.0	33.7	33.5	32.4	31.3	29.8	28.2
(1 + ΔN)W	-	32200	35300	37500	38200	38000	36400	35400	33800	30920
C_{LD}	.613	.676	.741	.788	.810	.810	.802	.807	.810	.782
C_L	.641	.680	.747	.772	.800	.793	.778	.800	.793	.788

NOTES:
1. γ Pitch = 4.3°/in. Start @ IAS = 100 Knots = 115.2 mph = 169.1 ft/sec
2. IAS = 8.43 ft/sec/in; TAS = 169.1 x 1.058 = 179.0 ft/sec
3. TAS = 1.058 x IAS = 8.93 ft/sec/in
4. W = 30433 lbs.; $\Delta N = \frac{.258}{1.35} = .1912$ G/IN

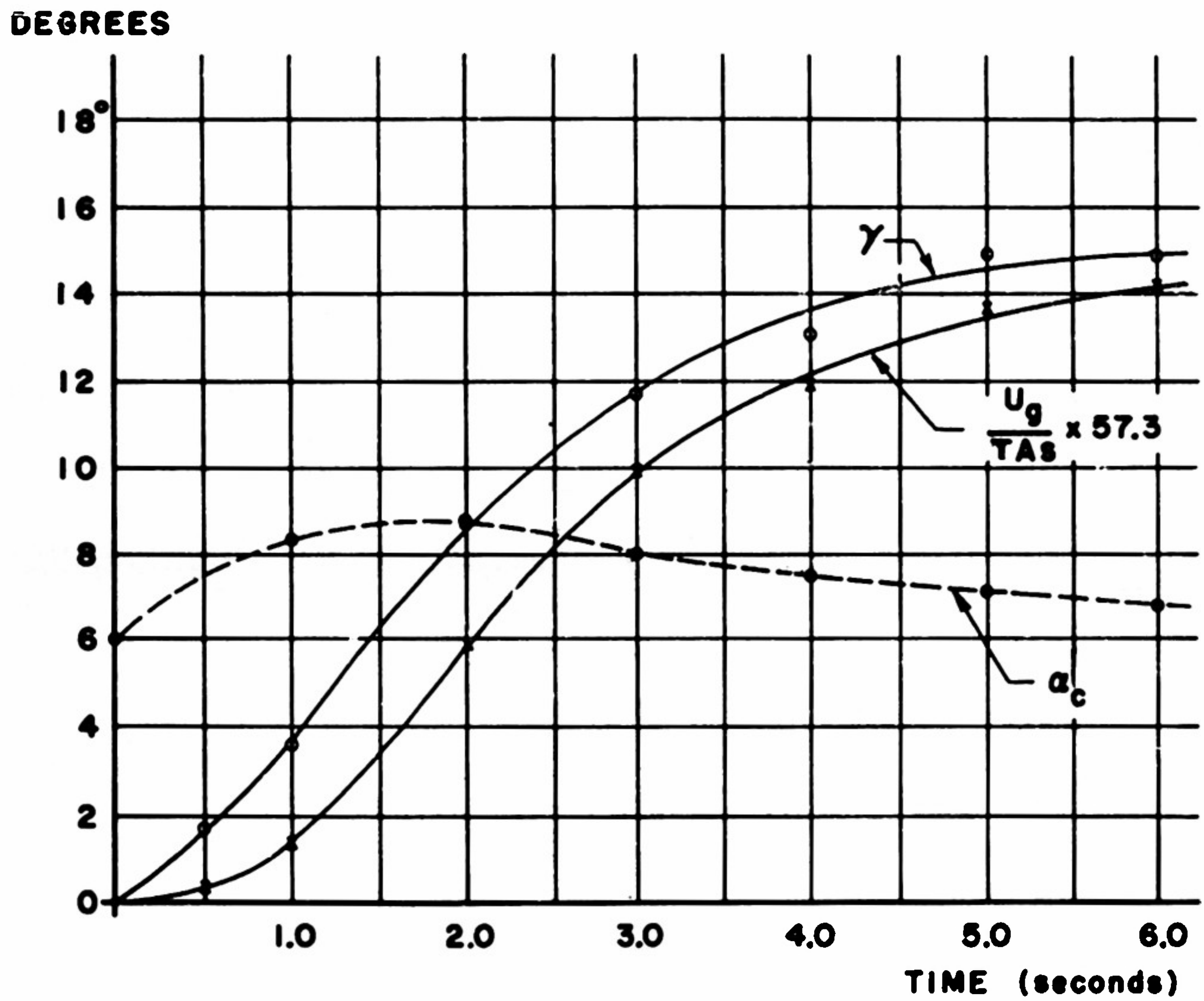

FIGURE 2. PITCH, RELATIVE WIND ANGLE AND CHORD PLANE ANGLE OF ATTACK vs. TIME, PULL-UP RUN 1965.

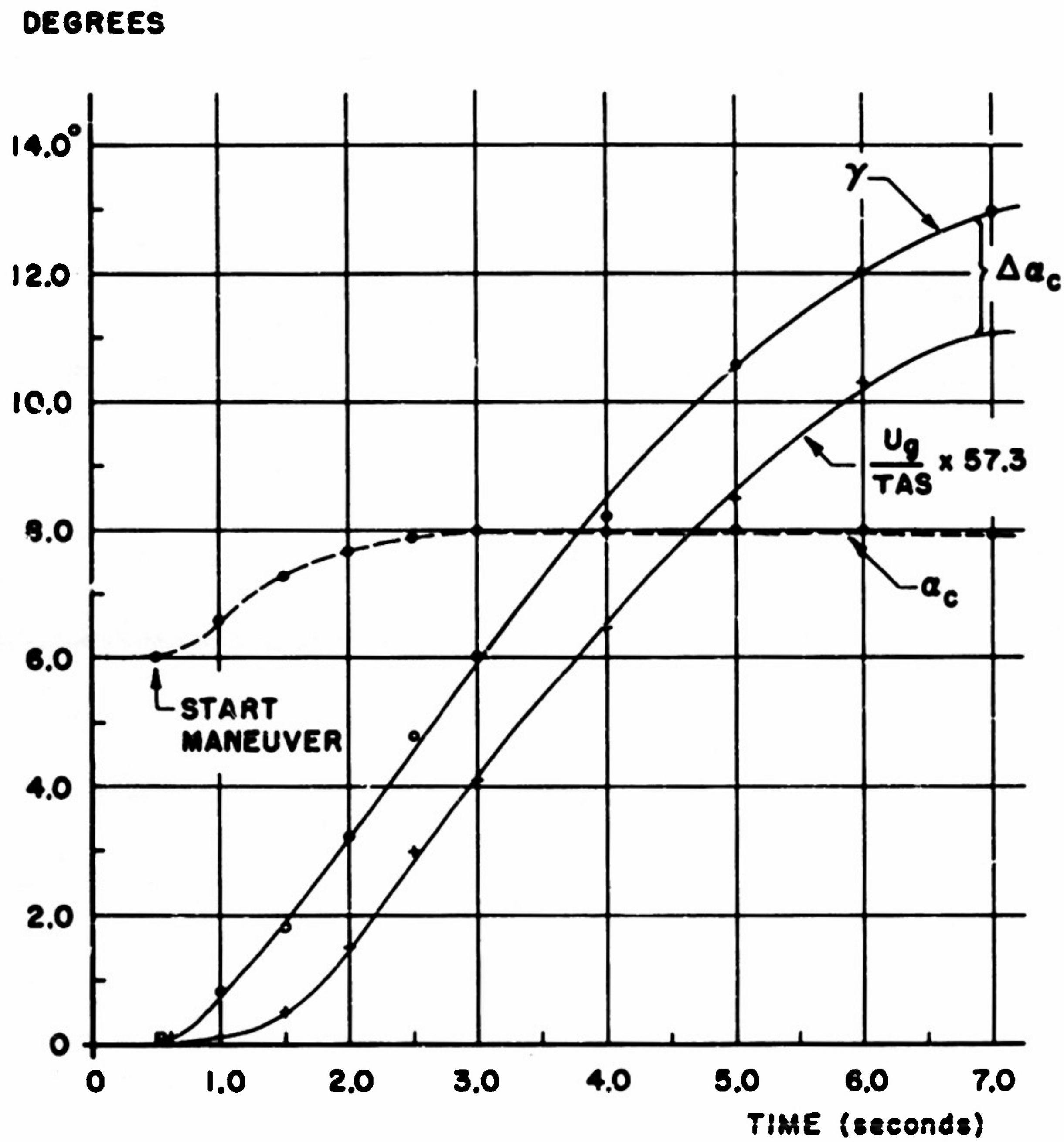

FIGURE 3. PITCH, RELATIVE WIND ANGLE AND CHORD PLANE ANGLE OF ATTACK vs. TIME, PULL-UP RUN 1969.

STRAIN GAGE ALTIMETER

An electric altimeter was built so that it could be used as an over-all check on the validity of the accelerometer predictions. A five element sylphon bellows was mechanically connected to a light Dural spring so that extension or contraction of the bellows would bend the spring. Four electric strain gages of the A-7 type were cemented to the spring, wired in bridge array and after drying, covered with neoprene. The entire assembly was then placed in a pressure tight case having a rubber pneumatic connection for attachment to the aircraft's static line: see Figure 3, Appendix No. 1.

The altimeter was connected to a flask and vacuum pump so that pressure might be lowered incrementally. A standard Kollsman aircraft altimeter was also connected to the flask and a Baldwin SR-4 Strain Indicator electrically connected to the altimeter. A small motor with an eccentric weight was fastened to the altimeter so that the vibration of the motor would break friction locks between the bellows and spring. A photograph of the altimeter calibration test is carried as Figure 4 in the Appendix No. 1. With the altimeter connected to the pneumatic and electronic apparatus as described, the pressure was lowered incrementally until a simulated altitude of 10,000 ft had been reached. Pressures at various altitudes were lowered quickly to detect hysteresis. The calibration test data are carried in the Table V and plotted on Figures 4 and 5.

TABLE V

CALIBRATING STRAIN GAGE ALTIMETER

Time	H_{press}	$H_{alt.}$	Hg_{true}	e_{up}	e_{down}
4:08PM	0	0	29.92"	0	-
4:09	1000	1000	28.86	465	455
4:10	2000	2000	27.82	915	910
4:12	3000	3000	26.81	1352	-
4:13½	4000	4000	25.84	1790	1780
4:15½	5000	5000	24.89	2190	-
4:18	6000	6000	23.98	2600	2590
4:19	7000	7000	23.09	3006	-
4:20	8000	8000	22.22	3421	3412
4:21	9000	9000	21.38	3792	-
4:22	10000	10000	20.58	4175	4169
4:23	9000	9000	21.38	-	3758
4:24	8000	8000	22.22	3345	3341
4:25	7000	7000	23.09	2938	-
4:26	6000	6000	23.98	2522	-
4:27	5000	5000	24.89	2100	-
4:27½	4000	4000	25.84	-	1675
4:28	3000	3000	26.81	-	1242
4:28½	2000	2000	27.82	-	809
4:29	1000	1000	28.86	-	366
4:30	0	15	-	-	-80
4:31	0	10	-	-	-69
4:32	0	7	-	-	-50
4:33	0	0	29.92	-	-48
4:34	500	500	-	195	-
4:35	500	500	-	195	-
4:36	0	0	29.92	-	-50
4:38	0	0	29.92	-	-41
4:39	1000	1000	28.86	432	-
4:40	1000	(from 1050)	28.86	-	422
4:40½	1000	(from 950)	28.86	428	-
4:41	0	2	29.92	-	-50
4:42	0	0+	29.92	-	-42
4:43½	0	0	29.92	-	-39
4:46	0	0	29.92	-	-38
4:47	500	500	-	208	-
4:48	500	500	-	199	-
4:49	500	500	-	198	-
4:50	500	500	-	197	-
4:50	0	0+	29.92	-	-42
4:51	0	0	29.92	-	-40
4:53	0	0	29.92	-	-38
5:00PM	0	0	29.92	-	-32

NOTES:
1. Date test 1-29-53 WHOI
2. Ref. "Handbook of Meteorology", Berry. Bollay and Beers McGraw Hill Book Co. 1945 p.103
3. H_{alt} altitude from Kollsman Altimeter #Q41248 set at 29.91, Barometer = 29.91" this date, 72°F inside
4. Flask in vacuum system, vibrating motor on altimeter
5. e = output strain micro in/in

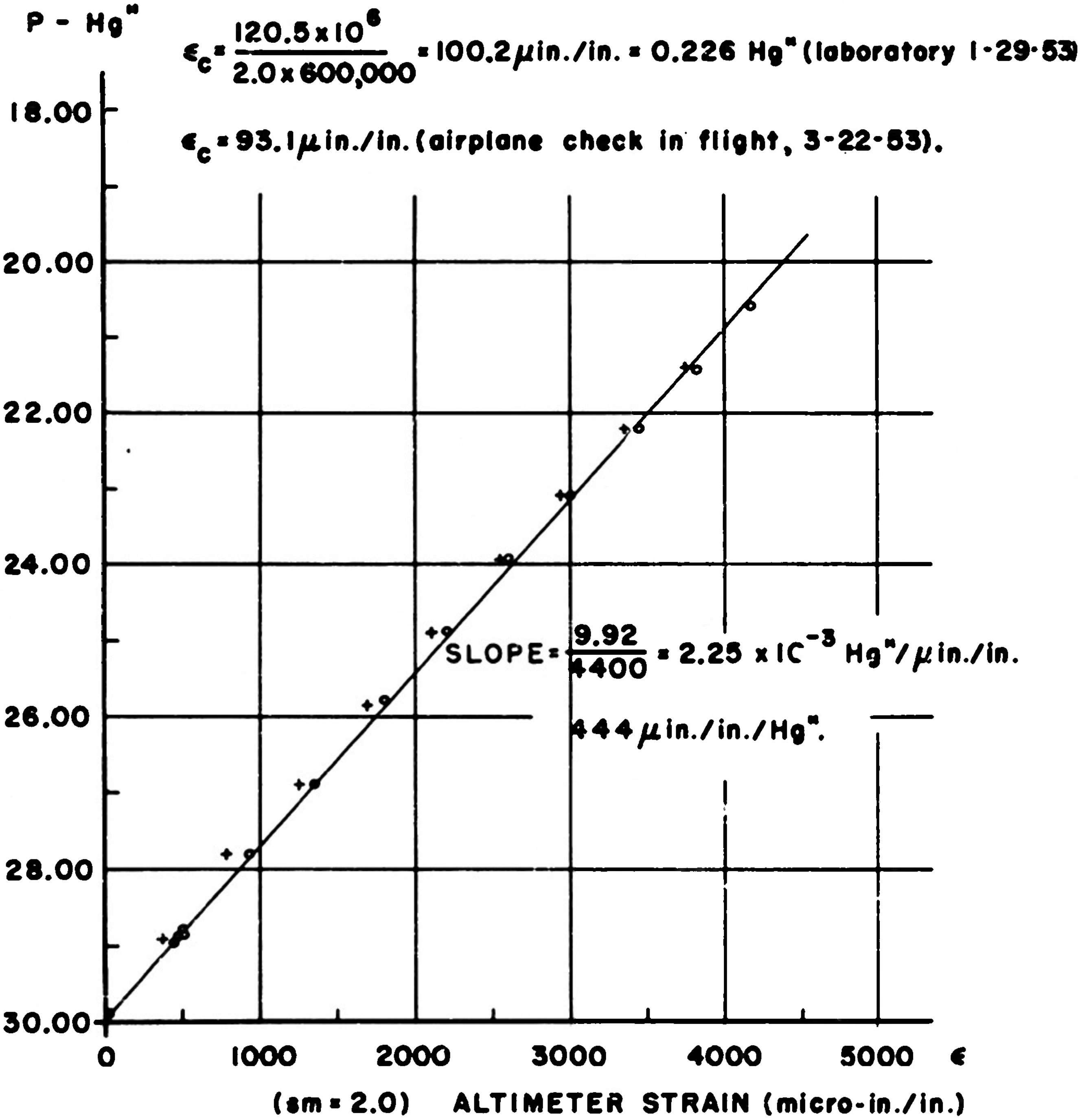

FIGURE 4. PLOT OF PRESSURE vs. STRAIN, ELECTRIC ALTIMETER.

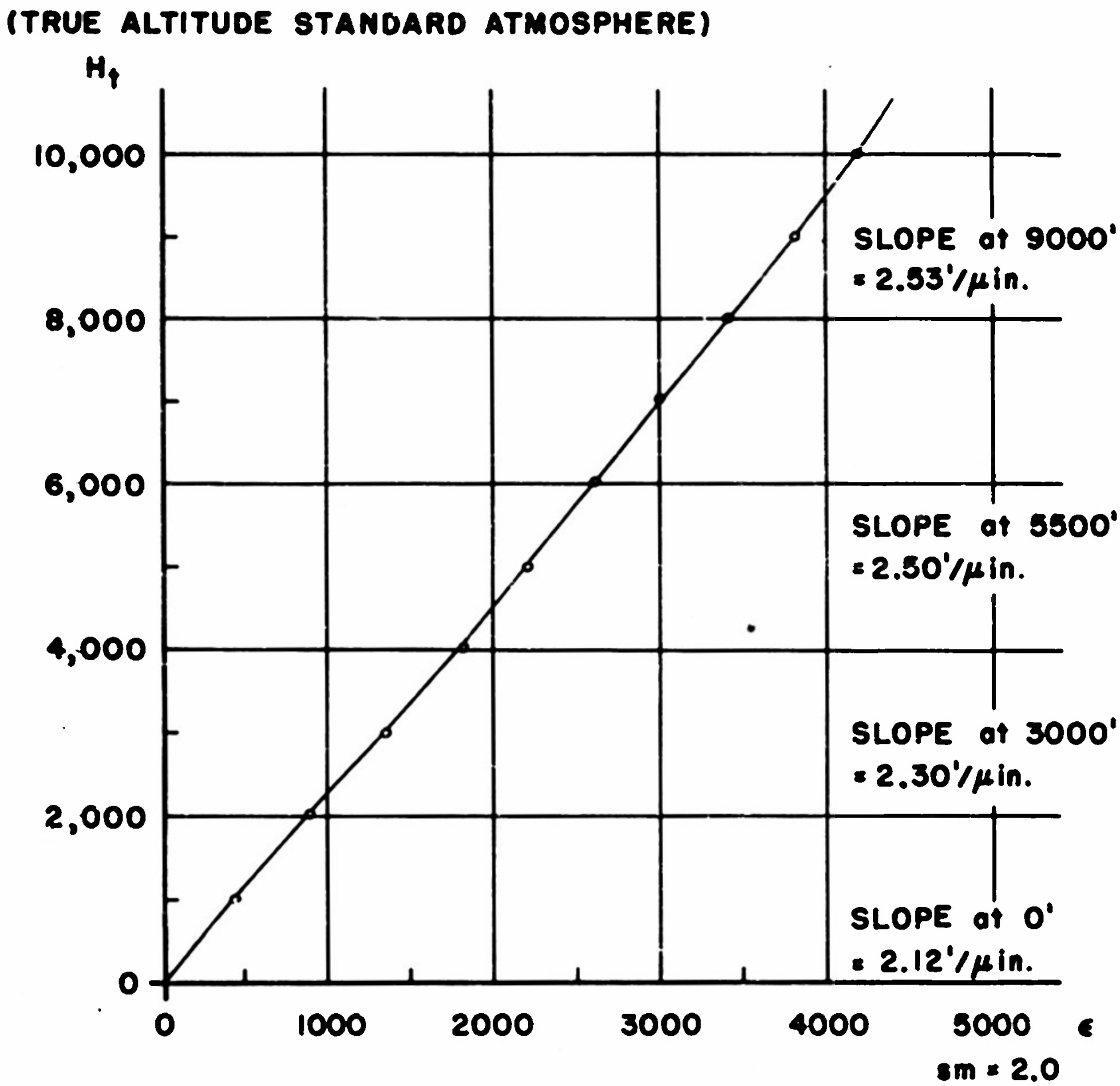

FIGURE 5. PRESSURE ALTITUDE vs. STRAIN OUTPUT, ELECTRIC ALTIMETER.

A shunting calibration resistor was placed across one bridge leg so that it could be used in flight to introduce a known equivalent altitude change. The sensitivity of the altimeter was checked in flight on a number of occasions and was found to be somewhat less, 7%, than predicted by laboratory tests. The decrease in sensitivity is attributed to the hardening of the neoprene in the Tropics, thereby increasing the spring constant of the strain gage element.

Early flight tests with the altimeter were performed using a static tube attached to the fuselage nose turret. It became quickly evident that serious fuselage interference existed at this point and that the static pressure varied seriously with altitude and speed: see Figure No. 6, Appendix No. 1. Before the flight testing off Puerto Rico in March of 1953, another pitot boom was obtained and attached to the right wing at outboard station 11 approximately 246" outboard from the airplane's centerline. The new boom placed the altimeter static holes 46" ahead of the wing leading edge or 0.27 of a chord length. In this location no interference effects were detected under steady flight at various attitudes or under severe turbulence. An apparent oscillation in the altimeter reading was detected under a very sharp pull-up. The frequency of oscillation was about 0.4 cps and is without explanation. Averaging out the oscillation observed brings the altimeter differential in agreement with the accelerometer integration. The natural frequency for the altimeter was found to be 120 cps.

As the altimeter case has appreciable volume, large pneumatic delays were encountered with the instrument mounted within the fuselage. For the 1953 Caribbean soundings, the altimeter was carried in the right wing with a short $\frac{1}{4}$" line connecting it to the pitot tube. Laboratory and field tests were conducted measuring the combined electrical and mechanical delay times of the altimeter and Statham anemometer. The delay data are carried in Table VI and indicate that the delay in response time for both pneumatic instruments is negligible.

TABLE VI

PNEUMATIC DELAY, ANEMOMETER AND ALTIMETER

DATE	INSTRUMENT	LOCATION	LINE			RISE TIME SEC
			LENGTH FT.	TYPE	O.D.	
3-3-53	ANEMOMETER	W.H.O.I.	0.0	-	-	.01
3-3-53	ANEMOMETER	W.H.O.I.	58.0	Cu	1/4"	.04
3-3-53	ANEMOMETER	W.H.O.I.	58.0	Cu	1/4"	.05
3-3-53	ANEMOMETER	W.H.O.I.	58.0	Cu	1/4"	.05
3-3-53	ANEMOMETER	W.H.O.I.	58.0	Cu	1/4"	.05
3-3-53	ALTIMETER	W.H.O.I.	10.0	P	1/4"	.14
3-3-53	ALTIMETER	W.H.O.I.	10.0	P	1/4"	.16
3-3-53	ALTIMETER	W.H.O.I.	58.0	Cu	1/4"	.50
3-3-53	ALTIMETER	W.H.O.I.	58.0	Cu	1/4"	.52
3-21-53	ANEMOMETER	PBY WING	4.5	Cu	1/4"*	.01
3-21-53	ALTIMETER	PBY WING	4.5	Cu	1/4"	.03

NOTES:

1. ANEMOMETER STATHAM P5-0.5-D-335
2. ALTIMETER, 4 Sr-4 strain gages on Dural spring. 120 Ω
3. GALVANOMETERS looking into 350 Ω , laboratory tests
4. Line code: Cu = copper; 1/4" Standard Tubing, 1/32" wall; P = Plastic
5. *release pressure on static holes Pitot tube, altimeter; dynamic hole and drain anemometer
6. Rise Time = time for 95% of flat top. after pressure release
7. 3-21-53 Tests with electrical and pneumatic constants same as in flight tests

STATHAM ANEMOMETER

As described in the Introduction, it is necessary to measure instantaneous vector air speed as part of the general method of gust determination. A Statham differential pressure gage was obtained and mounted on different occasions within the right wing or within the fuselage. This anemometer was calibrated against the two air-speed indicators of the PBY with the data carried on Table VII and plotted in Figure 6. Over the speed range from 90 to 120 knots the relationship between the aircraft's air-speed meters and the Statham pressure anemometer is seen to be linear. This speed change corresponds to an attitude change of about 5°.

As in the case of the altimeter, a calibration resistor was related to air speed changes so that it could be used as a continuous reference during flight testing.

On April 2, 1953, the PBY was checked by a ground speed run over the new Puerto Rico International Airport. The main landing strip is 7800 ft plus or minus 10 ft according to Mr. E. F. Heist, CAA District Airport Engineer. At 1000 ft altitude, temperature 25°C, upwind and downwind runs were made over the strip with the traverse time sighted through the drift sight and recorded by stop watch. The upwind run required 47 seconds and the downwind 37 seconds. The wind was E directly down the runway and reported as 19 mph. The aircraft was held between 103 and 104 knots on both runs or 107 to 108 TAS; measured ground speed was 109.7 knots, a discrepancy of 2.0%.

TABLE VII

CALIBRATION STATHAM ANEMOMETER - P5-0.5D-335 No. 391

DATE P_o Hg"	V_1 knots	TIME (PM)	H_p ft	t °C	W lbs	Cg inches	a_c deg.	ΔDEF in	e	Pitot tube
3-18-53	88	3:51	7000	14.5	29173	244.2	8.0	0	0	R.H
29.98	100						6.0	2.0	715	"
	110						5.0	3.9	1395	"
	120						3.0	5.9	2110	"
600KΩ Calib.		-	-	-	-	-	-	.78	279	"
3-21-53	90	2:34	6100	10.0	29923	249.3	-	0	0	"
30.05	100	2:35			29923	249.3	-	1.85	698	"
	110						-	3.50	1320	"
	110						4.5	3.55	1340	"
600KΩ Calib.		2:38	-	-	-	-	-	.74	279	"
4-2-53	80	3:39	6700	17.0	29908	250.5	-	0	0	L.H
29.94	90						-	1.1	458	"
	100						-	2.65	1104	"
	110						4.0	4.20	1750	"
600KΩ Calib.		-	-	-	-	-	-	.67	279	"
	90						7.5	1.25	521	"
	100						6.0	2.70	1124	"
	80						9.5	0	0	"
4-5-53	120-110	2:04	10500	10.0	29231	250.0	-	-1.7	-708	"
30.07	120-102						-	-2.9	-1208	"
	80						11.0	0	0	"
	90						8.0	1.35	580	"
	100						5.5	2.80	1202	"
	110						4.0	4.40	1890	"
600KΩ Calib.		-	-	-	-	-	-	.65	279	"

NOTES:
1. All tests using 5-116 oscillograph. 1 Kc amplifier
2. $e = 335 \times 10^6 / 600000 \times 2.0 = 279$ micro in/in
3. All tests in flight PBY-6A Navy 46683
4. Right hand pitot tube type AN-5816-2; Straight. Mounted at wing station 11, 247" outboard centerline, 42 inches ahead of wing leading edge. Chord at this station = 180"

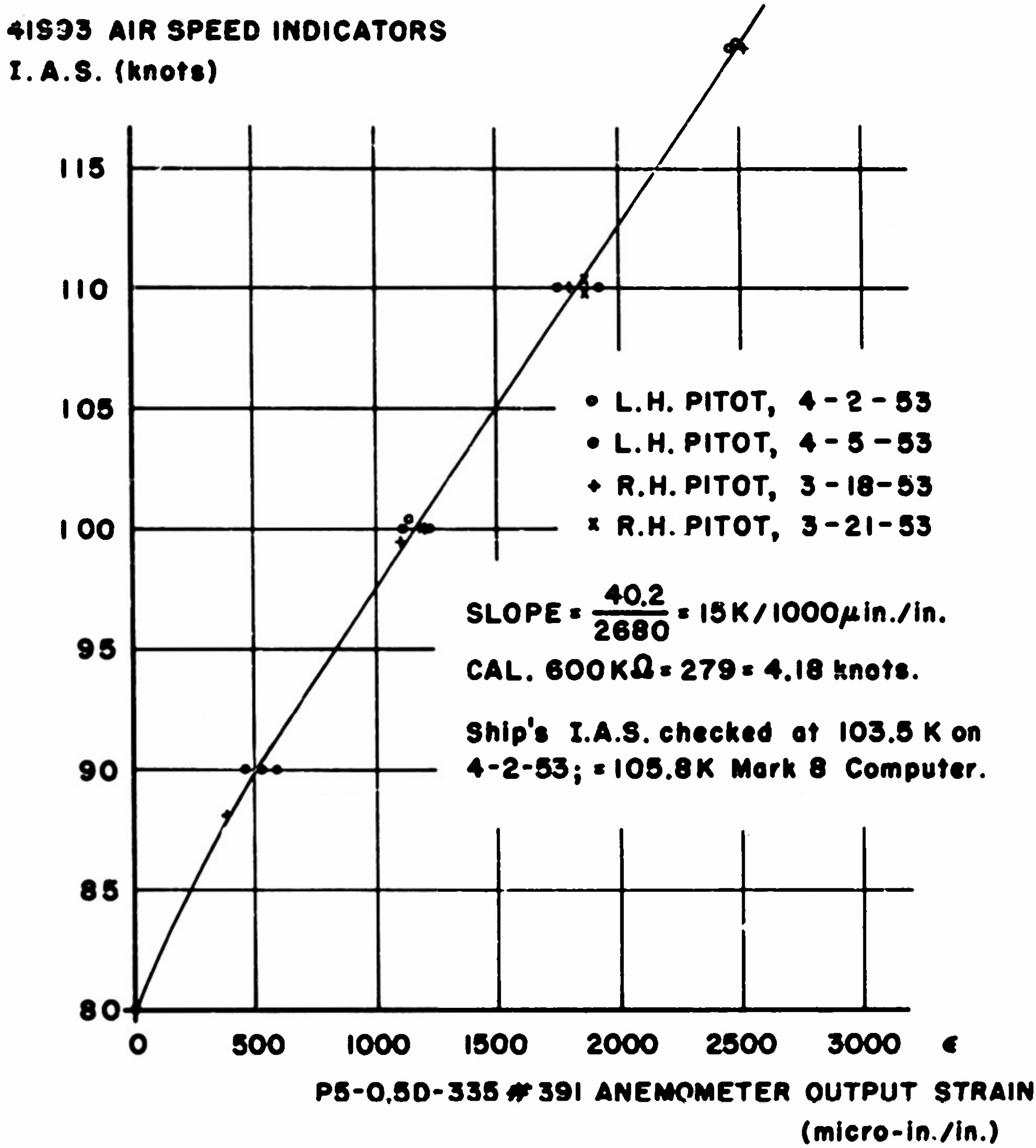

FIGURE 6. INDICATED AIR SPEED vs. ANEMOMETER OUTPUT, PBY-6A.

GYRO PITCH INDICATOR

In the general reconstruction of the aerodynamic forces acting upon the wing, it is necessary to know the instantaneous attitude of the airplane. An electric powered gyroscope was obtained and suitably connected to one channel of the Consolidated 5-116 oscillograph: see Figure 7. A 200 ohm potentiometer was connected into the gyro tilt circuit so that the galvanometer trace could be adjusted to any desired position during flight. A calibration resistor was connected into the circuit so that a displacement of a known amount of pitch could be recorded oscillographically during flight.

With the gyro electrically connected to the oscillograph, the gyro was tilted in increments up to 15°. The tilt angle was measured by means of a spirit level protractor. The relationship between galvanometer displacement and tilt angle is carried in the Table VIII and plotted on the Figure 8. It may be seen from Figure 8 that gyro output is linear up to at least ± 15°. Normally the pitch trace is carried at a screen position of 2.0" which is 1.0" above the bottom of the oscillograph paper. As the calibration resistor is little larger than the gyro bridge resistance, the calibration resistor will not cause the same displacements if the bridge is electrically moved very far from its zero position. A test was performed within the PBY measuring the variations in calibration displacement as a function of zero position: see Figure 9 and the Table IX. From Figure 9 it appears that the gyro trace may be moved ± ¼" or ± 2½° before serious error is introduced into the calibration system.

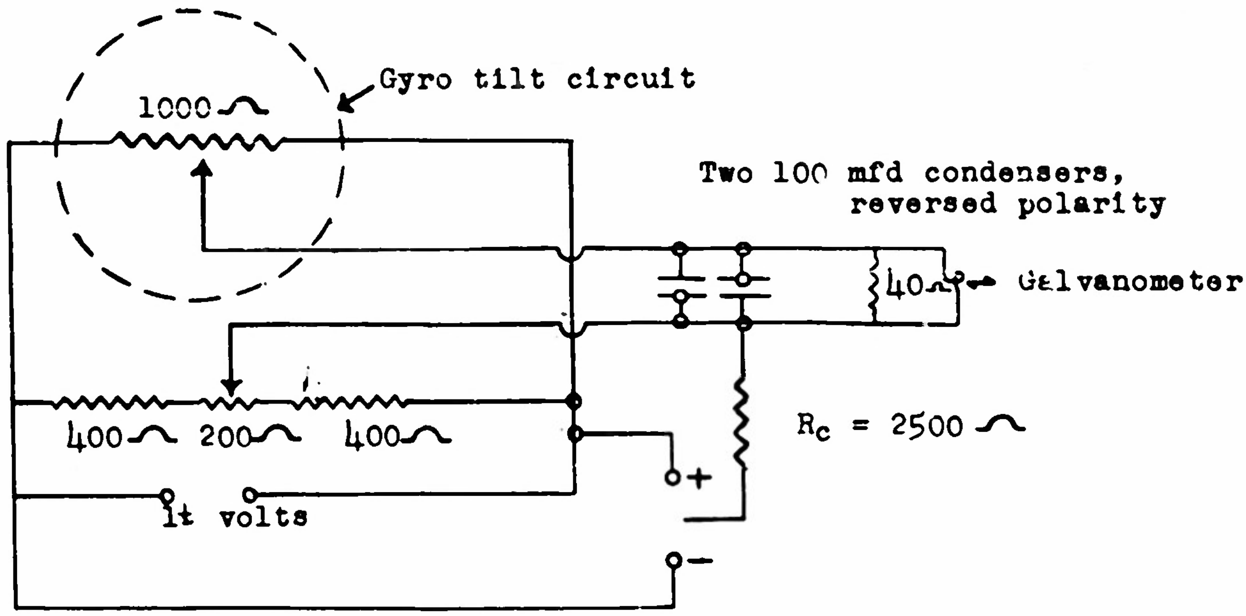

FIGURE 7. SCHEMATIC CIRCUIT OF C-1 PITCH GYRO

TABLE VIII

OUTPUT VS TILT, C-1 GYRO

TIME	TILTING GYRO	SCREEN DEF.
5:30PM	Angle	
5:40	0°	0*
	-5°	1.00"
	-10°	2.15"
5:44	-15°	3.35"
	0°	0**
	5°	-1.20"
	10°	-2.30"
	15°	-3.40"
6:02	0°	0 (± .05")

NOTES:

1. 1½ volt supply ± 1% from voltage regulated rectified AC source
2. U.S. AAF Type C-1 Autopilot Vertical Flight Gyro. Minneapolis Honeywell 26 volt dc
3. * Moved zero -1.0". ** Moved zero +2.0"
4. Calib. 2500 Ω = ± 1.15"
5. Test @ WHOI 3-3-53

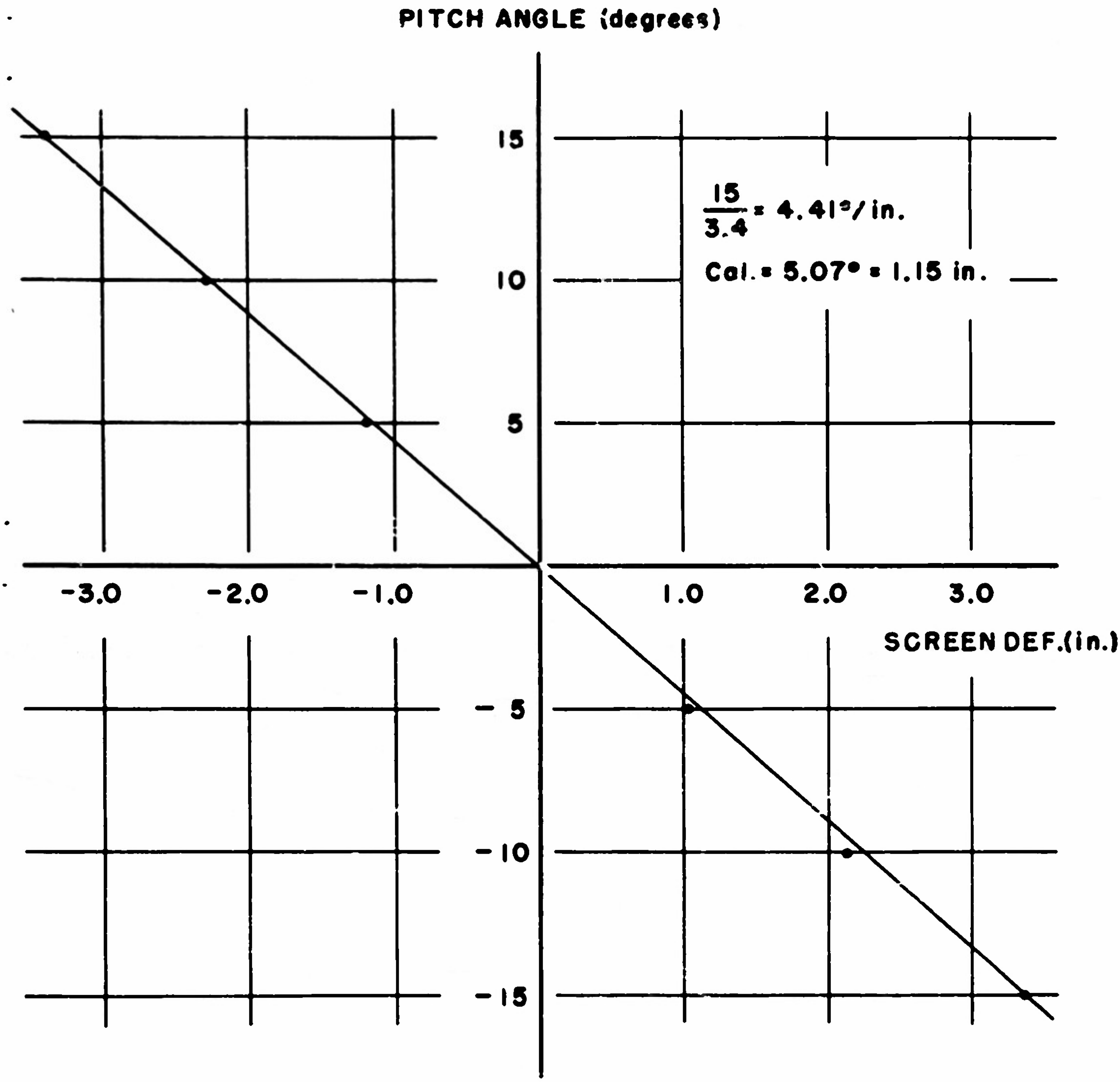

FIGURE 8. CALIBRATING C-1 GYRO IN LABORATORY ON 5-116 SCREEN.

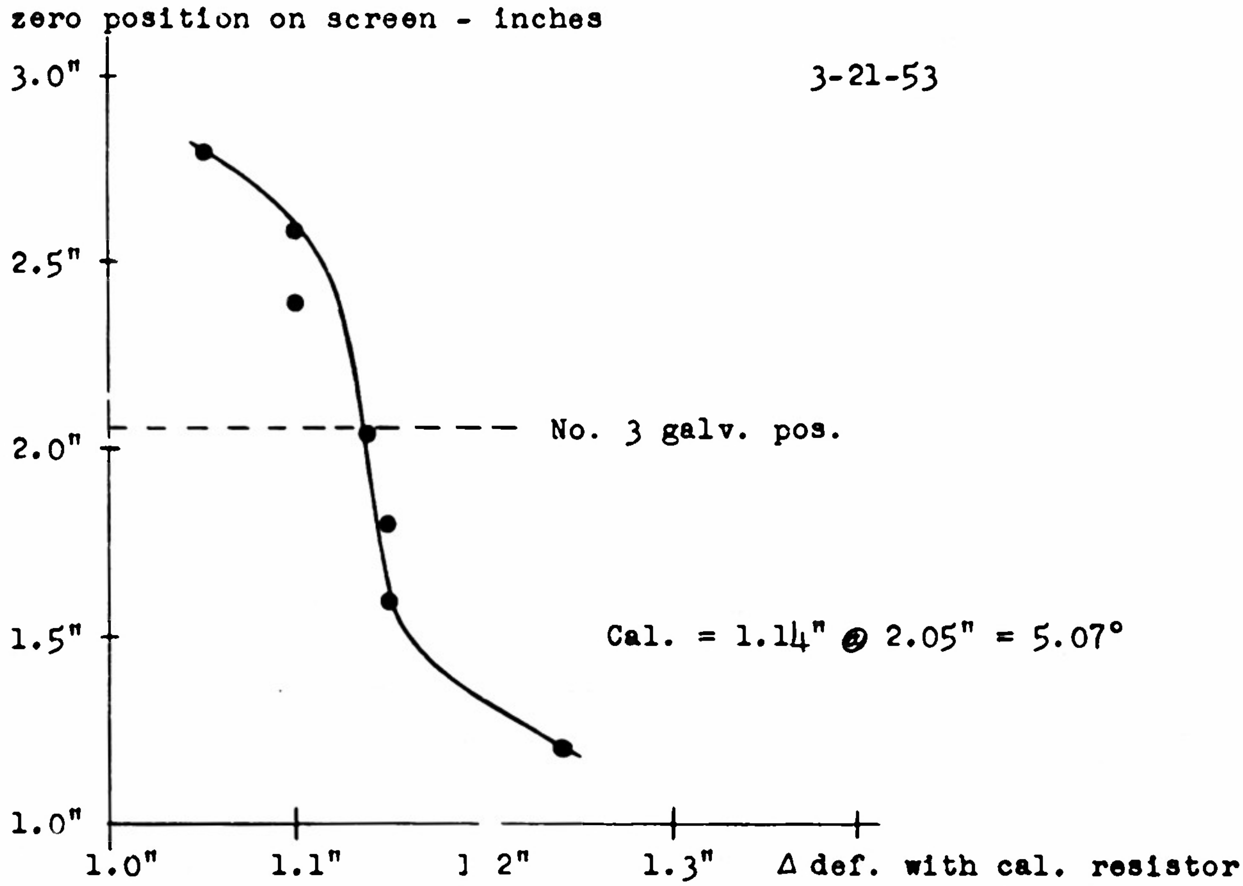

FIGURE 9. VARIATION IN CALIBRATION DISPLACEMENT WITH ZERO POSITION OF SPOT

TABLE IX

VARIATIONS IN CALIBRATION DEFLECTION WITH ZERO POSITION

ZERO POSITION*	+5.07° CALIB.	ΔDEF.
2.05	3.19"	1.14"
2.40	3.50	1.10
2.60	3.70	1.10
2.80	3.85	1.05
1.80	2.95	1.15
1.60	2.75	1.15
1.20	2.44	1.24
1.00	2.25	1.25

NOTES: 1. Test in PBY @ 12:11 PM 3-21-53
2. * on 5-116 oscillograph screen (inches)
3. Gyro about parallel to chord plane $\alpha_c \cong 5°$ on ground

INSTRUMENTS

A Statham 4-G accelerometer[11] was used to observe fuselage accelerations. This instrument and its use in the determination of PBY-6A wing loads has been described in reference 7. The anemometer, accelerometer, and electric altimeter each comprise an electrical bridge, the output of which is linear with the phenomenon being measured. Each instrument was cabled into a 1 kc carrier amplifier providing amplification and rectification of the transducer signals. The 1 kc amplifier output was used to drive the galvanometers within a Consolidated type 5-116 recording oscillograph. The gyro pitch indicator was connected directly into the oscillograph as was a side marker. Five channels of information were recorded simultaneously as a function of time. The side marker was used to permit correlation between the oscillograph records and the records simultaneously recorded on other instruments, such as the psychrograph, water column accelerometer and nose camera. Certain sections of the oscillograph records are carried in Figures 7 and 8 of Appendix No. 1.

CALIBRATION MANEUVERS

To simulate encounter with a vertical gust having a parabolic velocity profile, the aircraft was subjected to a number of sharp pull-ups in still air. The anemometer, accelerometer, pitch transducer, and altimeter traces were oscillographically recorded during these maneuvers. From an analysis of these records the validity of the instrumentation system may be known. The first integration of the accelerometer trace vs time must everywhere be equal to the first derivative of the altimeter trace.

On January 21, 1953, the PBY executed a sharp pull-up, run no. 1773, the analysis of which is carried in Figure 10. The run no. 1773 shows good correlation between the velocities predicted from the altimeter and the accelerometer except for a phase displacement indicating lag in the altimeter penumatic system.

During January, 1953, tests the altimeter was connected to the ship's static line in company with the entire complement of flight instruments. The length of line from the pitot boom combined with the rather large plenum chamber represented by the cavities of all instruments, seemed responsible for the altimeter delay. During the March tests the altimeter was moved into the leading edge of the right wing and connected by a short line to the static holes which it shared only with the Statham anemometer. In the wing position the altimeter pneumatic response was increased at least by a factor of 17 as compared to its January fuselage location: see Table VI.

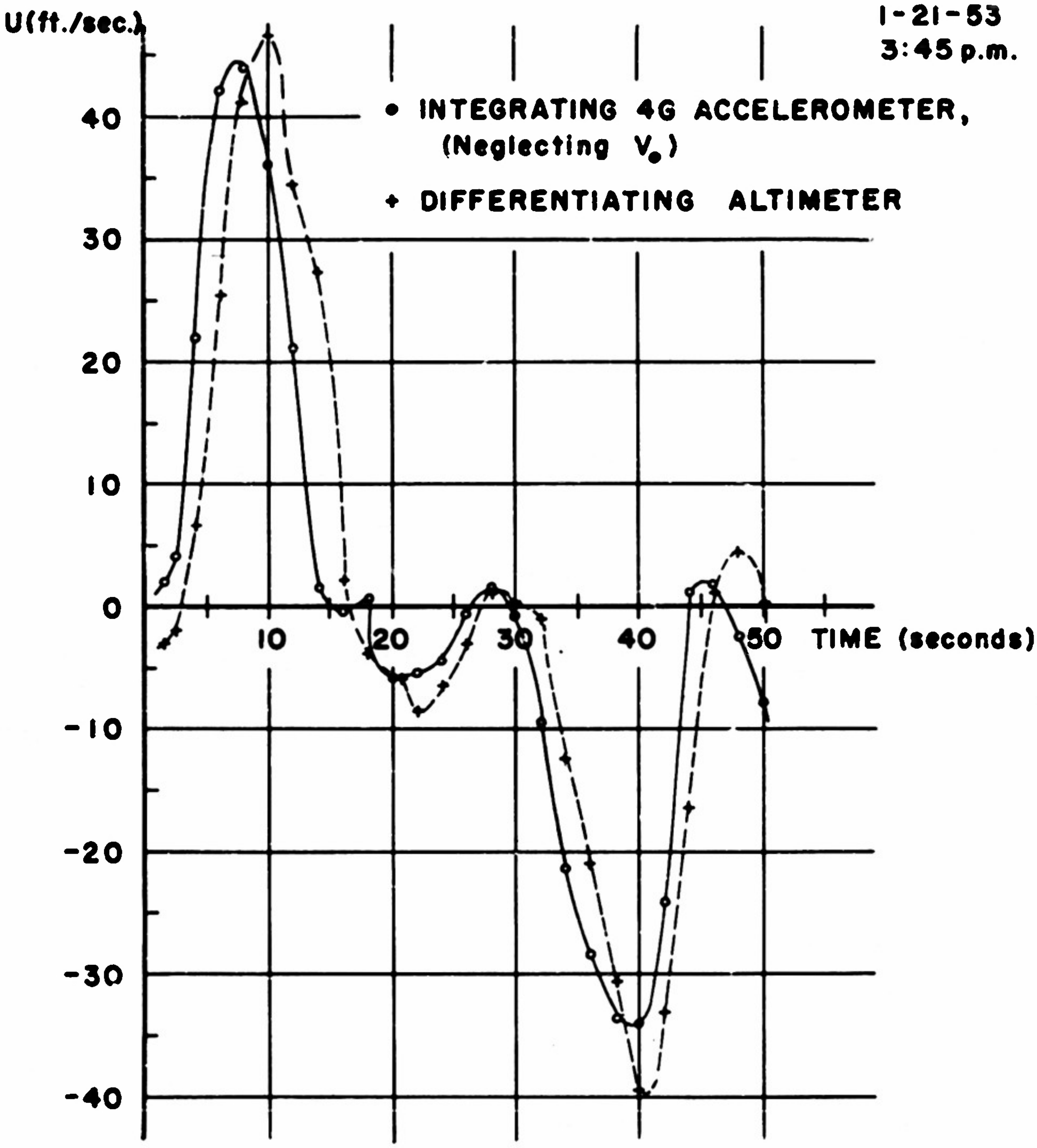

FIGURE 10. RUN 1773, CLIMB 300 ft. AT 600 ft./min., 1000' to 1300' AND BACK, ~ 115 knots.

On March 18, 1953, a number of pull-ups were executed and two records, nos. 1965 and 1969, have been analyzed. The data pertinent to analyses of the two runs are carried in Tables X, XI, and XII. Plots of the aircraft's vertical velocity as obtained from the accelerometer and altimeter are plotted together against time in the Figures 11 and 12. Correlation seems to be quite good for the run 1969. In the case of the very sharp pull-up run no. 1965, there appears to be an oscillation of very low frequency in the altimeter pneumatic system. This oscillation appears to be aerodynamic and as yet has no explanation. By averaging out the oscillation, its mean velocity agrees quite well with the same quantity predicted by the accelerometer.

For meteorological purposes it is desired that the airplane's instrumentation system respond to gust profiles of about 1.5 ft/sec^2 to 3 ft/sec^2. As the pull-up 1969 followed a rate of 7.14 ft/sec^2, it would seem that the instrumentation system was functioning satisfactorily.

TABLE X

PBY-6A Navy 46683

WEIGHT AND BALANCE 3-18-53

@ 2:30 PM Run 1969

ITEM	WEIGHT-lbs.	ARM-inches	WEIGHT X ARM
airplane	23,736	Cg = 252	5,996,982
gas (730 gals.)	4,380	267	1,170,000
oil (90 gals.)	675	208	140,400
Gingrass	180	113	20,340
Fournier	190	113	21,470
Kiernan	180	170.8	30,744
Rose	180	260	46,800
Bunker	180	170.8	30,744
Brewer	180	170.8	30,744
Malkus	130	346.8	45,084
Com. Savecker	156	346.8	54,101
McCasland	140	41.8	5,825
tool box	15	170.8	2,562
oscillograph	55	170.8	9,390
four ch. amplifier	30	170.8	5,120
Brewer inst. box	6	170.8	1,060
exp. pitot static tube	10	-36	-360
nose camera	10	0	0
	30,433 lbs.		7,611,006

$$Cg = \frac{7611006}{30433} = 250.0 \text{ in.}$$

TABLE XI

ANALYSIS OF SHARP PULL-UP, RUN NO. 1965

	ACCELEROMETER U_g = 12.69 FT/SEC/IN^2			ALTIMETER U_a = 37.15 TAN θ		
Time sec.	ΔA IN^2	ΔU_g	ΣU_g FT/SEC	θ	TAN θ	U_a FT/SEC
0.0	.00	.00	.00	.0	.000	.00
1.0	.34	4.32	4.32	29.5	.566	21.00
2.0	1.08	13.70	18.02	28.0	.532	19.78
3.0	.93	11.80	29.82	22.7	.418	15.56
4.0	.38	4.82	34.64	51.0	1.235	45.90
5.0	.31	3.94	38.58	42.5	.916	34.05
6.0	.00	.00	38.58	45.5	1.018	37.80
8.0	-.93	-11.80	26.78	32.0	.625	23.20
10.0	-1.43	-18.14	8.64	11.2	.198	7.35
11.0	-	-	-	9.0	.158	5.87
12.0	-1.00	-12.69	-4.05	-3.0	-.052	-1.93
14.0	-.31	-3.94	-7.99	-16.0	-.287	-10.67
15.0	-	-	-	-12.0	-.213	-7.91

NOTES: 1. Enter @ 100 Knots: pull-up 260 FT from 3000', 3-18-53

2.

Channel	No. 1	No. 2	No. 3	No. 4
Instrument	IAS	ACCEL.	PITCH	ALT
Calibrate 600k ⎍	.78" 7.08 FT/SEC	1.35" 8.31 FT/SEC^2	1.18" 5.07°	2.82" 218 FT (true)
Sensitivity	9.08 FT/SEC/IN	6.15 FT/SEC^2/IN	4.3 °/IN	77.3 FT/IN(true)

3. Check $3.5 \times 77.3 \times \frac{282.2}{287.5}$ = 266.2 indicated altitude change, from electric altimeter.

4. ΔA = planimetered area of accelerometer trace

5. θ = slope of altimeter trace

TABLE XII

ANALYSIS OF SHARP PULL-UP, RUN NO. 1969

	ACCELEROMETER U_g = 12.69 FT/SEC/IN^2			ALTIMETER U_a = 43.9 TAN Θ		
Time sec.	ΔA IN^2	ΔU_g	ΣU_g FT/SEC	Θ	TAN Θ	U_a FT/SEC
0.5	.00	.00	.00	.0	.000	.00
1.0	.02	.25	.25	5.0	.087	3.82
1.5	.11	1.40	1.65	9.0	.158	6.93
2.0	.24	3.05	4.70	13.5	.240	10.52
2.5	.32	4.06	8.76	17.0	.306	13.41
3.0	.30	3.81	12.57	19.5	.354	15.52
4.0	.56	7.11	19.68	26.0	.488	21.40
5.0	.44	5.58	25.26	32.0	.625	27.40
6.0	.34	4.31	29.57	34.5	.687	30.20
7.0	.18	2.28	31.85	36.5	.739	32.40
8.0	-.10	-1.27	30.58	36.5	.739	32.40
9.0	-.17	-2.06	28.52	35.0	.700	30.70
10.0	-.23	-2.92	25.60	28.0	.532	23.38
11.0	-.47	-5.96	19.64	22.0	.404	17.72
12.0	-.60	-7.61	12.03	18.0	.325	14.26
13.0	-.56	-7.11	4.92	9.0	.158	6.94
14.0	-.49	-6.21	-1.29	0.0	.000	0.00
15.0	-.36	-4.56	-5.85	-5.0	-.087	-3.82
16.0	-.18	-2.29	-8.14	-8.5	-.149	-6.55
17.0	-.09	-1.14	-9.28	-10.5	-.185	-8.13
18.0	-.03	-.38	-9.66	-12.0	-.213	-9.36
19.0	.07	.89	-8.77	-13.0	-.231	-10.15
20.0	.10	1.27	-7.50	-14.0	-.249	-10.94
21.0	.015	1.91	-5.59	-13.9	-.247	-10.85
22.0	-.14	-1.78	-7.37	-13.9	-.247	-10.85
23.0	-.23	-2.92	-10.29	-18.0	-.325	-14.30
24.0	-.245	-3.11	-13.40	-20.5	-.374	-16.43
25.0	-.180	-2.28	-15.68	-22.5	-.414	-18.20
26.0	-.170	-2.16	-17.84	-24.5	-.456	-20.02
27.0	-.045	-.67	-18.51	-23.0	-.424	-18.65
28.0	.085	1.08	-17.43	-23.5	-.435	-19.11
29.0	.195	2.47	-14.96	-21.5	-.394	-17.31
30.0	.160	2.03	-12.93	-18.7	-.338	-14.85
31.0	.140	1.78	-11.15	-15.5	-.277	-12.19
32.0	.170	2.16	-8.99	-14.0	-.249	-10.92

TABLE XII (Cont)

Time	ΔA	ΔU_g	ΣU_g	θ	TAN θ	U_a
33.0	.200	2.54	-6.45	-13.0	-.231	-10.15
34.0	.07	.89	-5.56	-9.5	-.167	-7.34
35.0	.05	.64	-4.92	-6.2	-.109	-4.79
36.0	.025	.32	-4.60	-6.0	-.105	-4.61
37.0	.02	.25	-4.35	-5.5	-.096	-4.22
38.0	.10	1.27	-3.08	-4.5	-.079	-3.47
39.0	.095	1.21	-1.87	-5.0	-.087	-3.82
40.0	.105	1.33	-.54	-4.0	-.070	-3.07
41.0	.01	.13	-.41	-2.8	-.049	-2.15
42.0	-.025	-.32	-.73	0.0	.000	0.00

NOTES: 1. Altitude 3000 ft. Barometer at sea level = 29.97 Hg"
Temperature at 3000 ft. = 14.5°C. T_{std} = 9.2°C
Airplane gross weight = 30433 lbs. Cg @ 250.0 inches.
Time: 2:30 PM

2.

Channel	No. 1	No. 2	No. 3	No. 4
Instrument	IAS	ACCEL.	PITCH	ALT
Calibrate 600k ⎍	.84" 4.18 Knots 7.08 ft/sec	-1.35" .258 G's 8.31 ft/sec^2	1.18" 5.07°	2.41" 218 FT (true)
Sensitivity	8.43 ft/sec/in	6.15 ft/sec^2/in	4.30 °/in	90.5 ft/in

3. Check $2.95 \times 90.5 \times \frac{282.2}{287.5}$ = 263.0 ft. indicated altitude change from electric altimeter.
Observed gain on airplane altimeter = 265 ft.

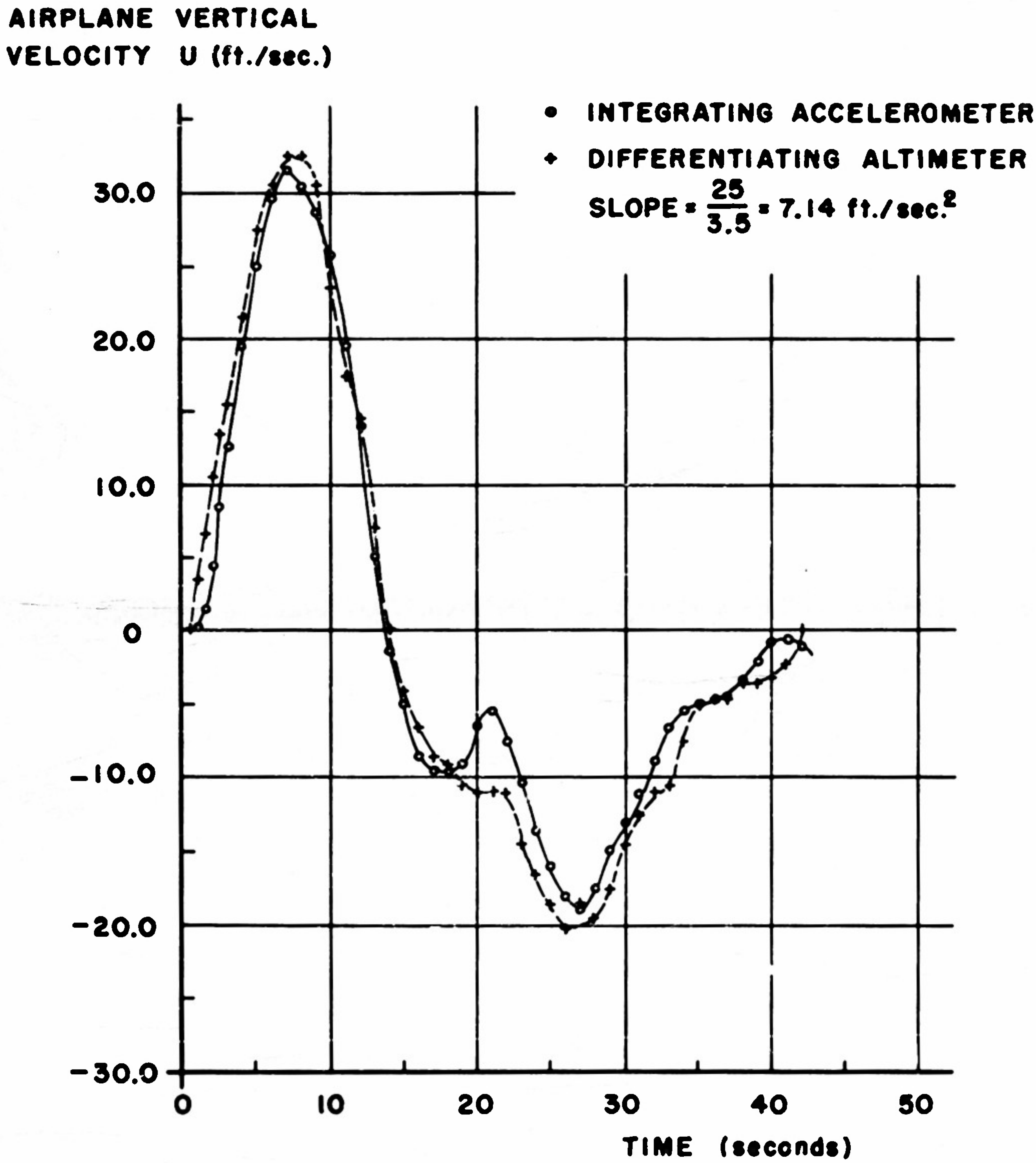

FIGURE 11. RUN 1969; 3-18-53 at 2:30 p.m. H_p = 3000 ft.

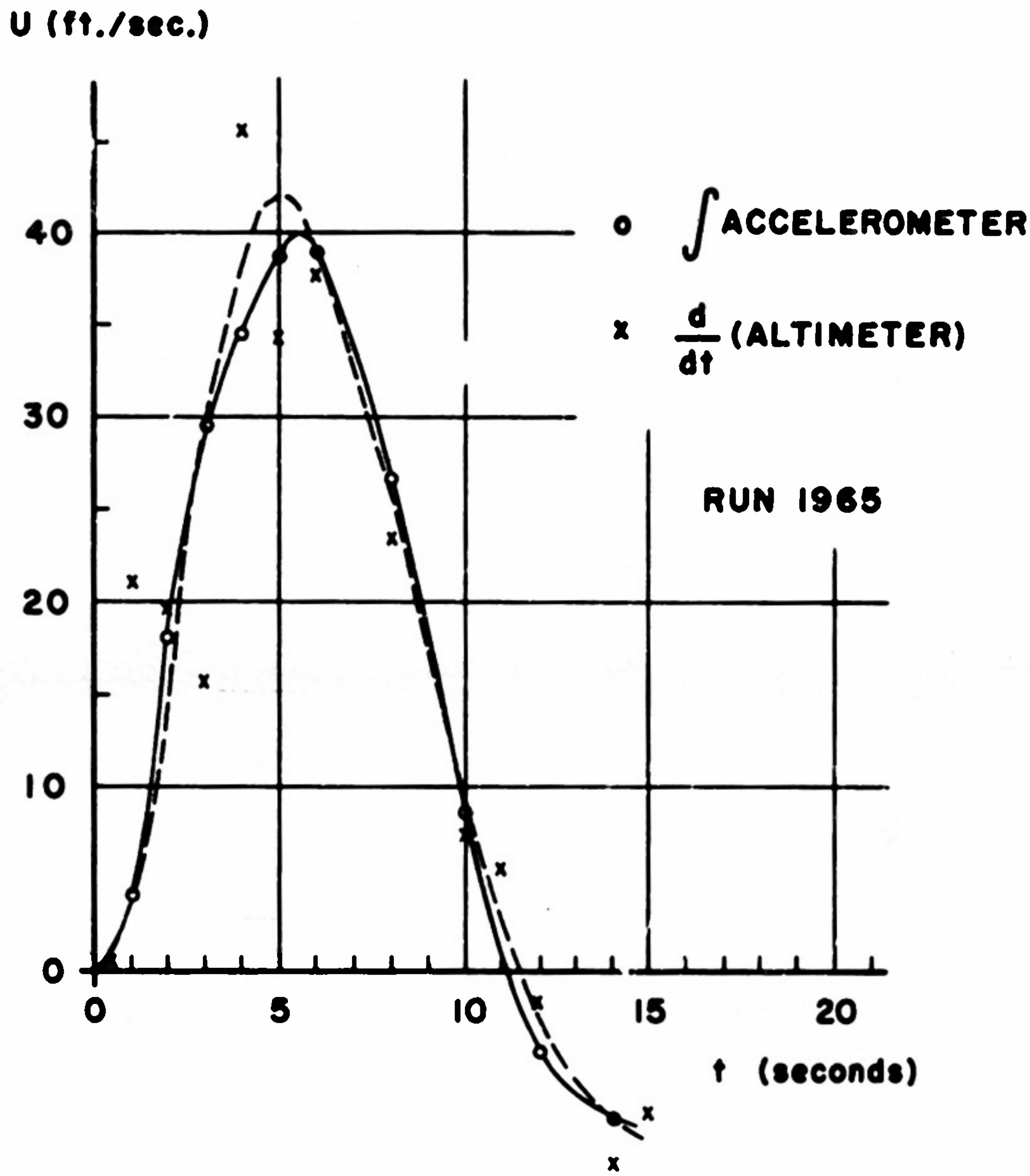

FIGURE 12. PLOT OF AIRPLANE VERTICAL VELOCITY vs. TIME, PULL-UP RUN 1965.

WING WHIP

On April 27, 1953, a 4-G accelerometer was attached to the left-hand wing float so that the fundamental bending frequency of the wing might be determined. With the aircraft resting on its three wheels, the left wing float was jacked upward until a force of a few hundred pounds had been developed. A wooden link between the jack and the float was then knocked out permitting the wing to restore to its equilibrium position through a series of damped oscillations. The wing oscillations were observed oscillographically and, upon development, the natural frequency of the wing was found to be about 3¼ cps. This procedure was reviewed by Professor J. P. Den Hartog of the Massachusetts Institute of Technology who stated that is was his belief that this experimental procedure was valid. Examination of the oscillographic records of lift strut strain gage records from the reference 7 confirm oscillations of small magnitude in flight of 3¼ cps.

In the reference 9 it has been shown that acceleration errors due to wing whip are negligible if the natural frequency of the wing is 5 times that of the gust frequency, assuming the latter to be a sinusoid. Upon this basis[9] the PBY-6A should behave as a rigid body encountering gusts of 0.7 cps or lower in frequency; however, examination of the records in reference 7 indicates correlation between lift strut strains and accelerations at 2 cps. In a strut braced wing the engine and fuel accelerations are of opposite sign to the outer panel wing whip accelerations and therefore, to an extent cancel. This cancellation does seem to take place as

frequencies of 3.5 cps observed in the wing strut strain gages do not exist in the fuselage accelerometer traces: see end of run 1218, ref. 7. Frequencies of about 2.0 cps are found in the same magnitude in both accelerometer and strut strain gage traces. It is concluded, then, that the PBY-6A fuselage accelerometers will follow gusts of at least 0.7 cps, excluding, of course, effects of circulation lag.

ANGULAR ACCELERATION

Encountering turbulence, the aircraft changes pitch attitude under accelerated loading; consequently, fuselage accelerations, as observed, will combine the two components of translation and rotation. The aircraft rotational component will center about the Cg of the airplane mass and will increase linearly with radial distance outward. Only the translational acceleration component is of interest meteorologically and must be separated from rotational acceleration.

To evaluate accelerometer errors due to placement at a distance from the airplane Cg, three accelerometers were placed at various stations along the fuselage and the airplane flown through a series of maneuvers and turbulence. The data obtained are summarized in Table XIII and indicate that no appreciable error will accrue from placement within 50" of the airplane's horizontal Cg. The water column accelerometer placed 117" aft of the Cg shows little error under maneuver conditions and the discrepancies during turbulence are more probably due to pen overshoot rather than rotation, since the instrument read too high for positive accelerations.

TABLE XIII

AIRPLANE ACCELERATIONS AT VARIOUS STATIONS

RUN	12 G		4 G		WATER COLUMN	DESCRIPTION
	DEF(in)	G'S	DEF(in)	G'S	G'S	
1773	1.450	1.412	1.800	1.421	1.44	PULL UP
1773	-1.200	0.659	-1.470	0.656	0.67	ARREST CLIMB
1773	1.650	1.468	2.010	1.471	1.50	ARREST DESCENT
1774	.540	1.153	.630	1.148	1.20	SHARP
1774	-.600	0.830	-.720	0.832	0.84	TURBULENCE
1775	-.470	0.867	-.620	0.855	0.89	SHARP
1775	.560	1.159	.610	1.143	1.20	TURBULENCE
1775	.506	1.144	.625	1.146	1.20	TURBULENCE

NOTES: 1

ACCELEROMETER	SENSITIVITY	STATION
4 G	.234 G/IN	212"
12 G	.284 G/IN	305"
WATER COLUMN	.445 G/IN	367"

2. All tests PBY-6A 1-21-53
3. Pitch change was ~17° in 9 seconds, RUN 1773
4. Cg approx. station 250"

SLOW MANEUVERS

It is not possible to hold the aircraft at constant pitch attitude during cloud traversing, consequently, there is some velocity change due to changing pitch attitude, not due to gust forces directly. It was desired to determine the speed change of the airplane in smooth air resulting from small changes in pitch. It was hoped that from the oscillographic flight records a simplified mathematical notation might be evolved relating changes in pitch angle with forward velocity. With the relationship between pitch angle and speed established, it would, then, be possible to separate out of cloud run data the speed changes due solely to attitude change.

At first thought, it might seem that graphical methods used for take-off calculations might be used to predict speed changes in slides. However, the accelerating force due to pitch is known only as a function of time whereas drag is related to velocity. It was necessary to solve the problem analytically using average values of $W \sin\gamma$ and ΔC_d. These values of force and drag coefficient must be planimetered for each period of time chosen, making the solution very laborious. The mathematical concept assumes an accelerating force resulting from the airplane pitching at an angle γ from its equilibrium position; a change in drag as a function of the change in velocity squared $(V^2-V_o{}^2)$ and a decrease in thrust as a linear function of the velocity change (v_1/V_o). The mathematical treatment is presented in the following pages and has been solved for two cases: runs 2079 and 2080.

The oscillographic records for runs 2079 and 2080 have been analyzed in Tables XIV and XV and plotted on the Figures 13 and 14. As can be seen from the analytical examples, correlation between theory and measurement is not close and it may be necessary ultimately to make an emperical solution with a nomographic presentation.

TABLE XIV

SPEED CHANGE WITH PITCH, NOSE UP, RUN 2079

TIME	PITCH		IAS KNOTS		
sec.	DEF(in)	γ	DEF(in)	ΔK_1	K_1
0	.00	.000	.00	.000	100.00
1	.00	.000	.00	.000	100.00
2	.00	.000	.00	.000	100.00
3	.00	.000	.00	.000	100.00
4	.17	.684	.00	.000	100.00
5	.38	1.530	-.02	-.121	99.88
6	.52	2.090	-.12	-.728	99.27
7	.73	2.940	-.22	-1.330	98.67
8	.80	3.220	-.40	-2.420	97.58
10	.81	3.260	-.73	-4.420	95.58
12	.78	3.140	-1.02	-6.190	93.81
14	.70	2.820	-1.22	-7.390	92.61
16	.83	3.340	-1.44	-8.720	91.28
18	.85	3.420	-1.61	-9.750	90.25
20	.90	3.620	-1.77	-10.700	89.30
22	.95	3.820	-1.92	-11.600	88.40
27	.92	3.700	-2.31	-14.000	86.00
30	.68	2.730	-2.43	-14.750	85.25

NOTES: 1. IAS = .69" Screen Deflection = 4.18 K_1 = 6.06 K_1/in; Pitch = 1.26" = 5.07° = 4.02 °/in

2. Time 12:45 PM; G.W. = 29792 lbs; Altitude 6750 Ft. t = 13.7°C; P_o = 30.03 (Hg")

TABLE XV

SPEED CHANGE WITH PITCH, NOSE DOWN, RUN 2080

TIME	PITCH		IAS KNOTS		
sec.	DEF(in)	γ	DEF(in)	ΔK_1	K_1
0	.00	0.00	0.00	0.00	100.00
1	-.25	-1.01	0.00	0.00	100.00
2	-.47	-1.89	0.00	0.00	100.00
3	-.70	-2.82	0.00	0.00	100.00
4	-.60	-2.42	0.13	0.79	100.79
6	-.68	-2.74	0.36	2.18	102.20
8	-.65	-2.62	0.52	3.15	103.15
10	-.55	-2.22	0.66	4.00	104.00
12	-.80	-3.22	0.77	4.66	104.66
14	-.75	-3.02	1.00	6.06	106.06
16	-.65	-2.62	1.18	7.15	107.15
18	-.70	-2.82	1.28	7.75	107.75
20	-.73	-2.94	1.42	8.60	108.60
22	-.73	-2.94	1.60	9.69	109.69

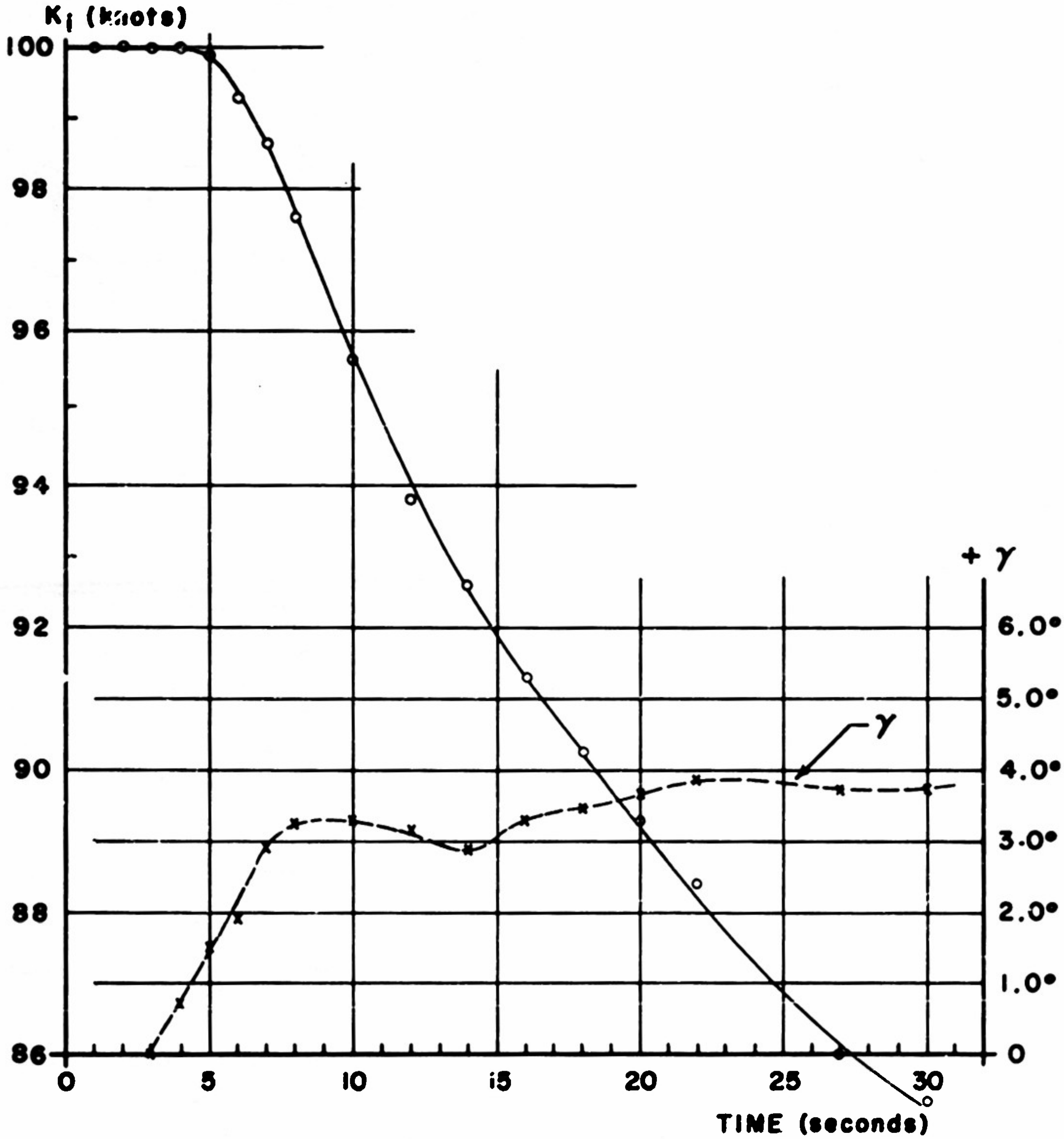

FIGURE 13. PLOT OF AIR SPEED AND PITCH ANGLE vs. TIME FOR SLOW NOSE UP RUN 2079.

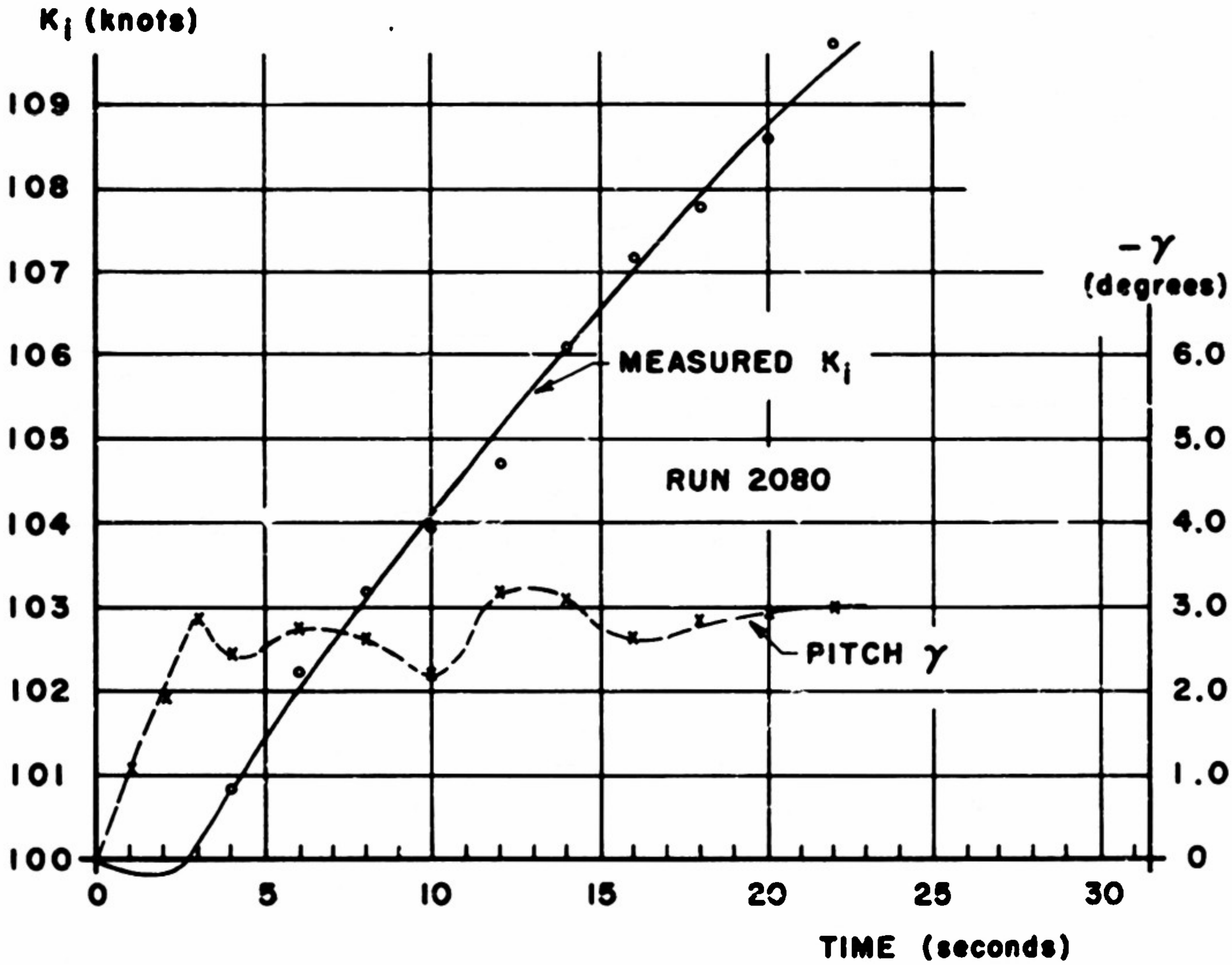

FIGURE 14. PLOT OF AIR SPEED vs. TIME FOR SLOW NOSE DOWN SLIDE RUN 2080.

DERIVATION

let: V = indicated air speed of airplane at any time t

V_o = indicated air speed of airplane at $t = 0$ (= 100 knots) = 169.3 ft/sec

t = time in seconds

v = speed change in time $t = V - V_o = v$

V_t = true air speed = $1.13\ V$ @ 6750' ; 13.7°C

γ = pitch angle change from equilibrium - degrees

$M = W/g = 29792/32.2 = 923$ - slugs

C_{d_o} = drag coefficient = .052

S = wing area = 1400 ft2

F = accelerating force - lbs

D = drag = $C_d \cdot S \cdot .001185 V^2$ - lbs; (V - ft/sec)

$\phi = W \sin \gamma$

$$F = W \sin \gamma + C_{d_o} \cdot q_o \cdot S - C_d \cdot q \cdot S - C_{d_o} \cdot q_o \cdot S \frac{v_1}{V_o}$$

now $C_d = (C_{d_o} + \Delta C_d)$; $\beta = .001185$

$$F = \phi + C_{d_o} \cdot \beta \cdot V_o^2 \cdot S - C_d \cdot \beta \cdot V^2 \cdot S - C_{d_o} \cdot \beta \cdot V_o^2 \frac{v_1}{V_o} \cdot S$$

$$F = \phi + C_{d_o} \cdot \beta \cdot V_o^2 \cdot S - C_{d_o} \cdot \beta \cdot V^2 \cdot S - \Delta C_d \cdot \beta \cdot V^2 \cdot S - C_{d_o} \cdot \beta \cdot V_o^2 \frac{v_1}{V_o} \cdot S$$

$$F = \phi + C_{d_o} \cdot \beta \cdot S (V_o^2 - V^2) - \Delta C_d \cdot \beta \cdot V^2 \cdot S - C_{d_o} \cdot \beta \cdot V_o \cdot v_1 \cdot S$$

$$F = \phi + C_{d_o} \cdot \beta \cdot S(-2V_o\ v_1) - \Delta C_d \cdot \beta \cdot S\ (V_o^2 + 2V_o\ v_1) - C_{d_o} \cdot \beta \cdot V_o \cdot S \cdot v_1$$

$$F = \phi + v_1 \left[-2V_o\ C_{d_o} \cdot \beta \cdot S - \Delta C_d \cdot \beta \cdot S \cdot 2 \cdot V_o - C_{d_o} \cdot S \cdot \beta \cdot V_o\right] - \Delta C_d \cdot \beta \cdot S \cdot V_o^2$$

let: $a = \emptyset - \Delta C_d \cdot \beta \cdot S \cdot V_o^2$

$$b = -2 \cdot V_o \cdot C_{d_o} \cdot \beta \cdot S - \Delta C_d \cdot \beta \cdot S \cdot 2 \cdot V_o - C_{d_o} \cdot S \cdot \beta \cdot V_o = - S \cdot \beta \cdot V_o (3C_{d_o} + 2\Delta C_d)$$

then: $F = M \frac{dv}{dt}$

$$dt = \frac{M\,dv}{F} = \frac{M\,1.13\,dv_1}{a + bv_1}$$

Integrating: $t \Big/_0^t = \frac{1.13M}{b} \ln(a + bv_1) \Big/_0^{v_1}$

$$t = \frac{1.13M}{b} \ln\left(\frac{a + bv_1}{a}\right)$$

$$\frac{bt}{1.13M} = \ln\left(\frac{a + bv_1}{a}\right)$$

$$e^{\frac{bt}{1.13M}} = \frac{a + bv_1}{a}$$

$$\frac{a \cdot e^{\frac{bt}{1.13M}} - a}{b} = v_1$$

$$\frac{a}{b}\left(e^{\frac{bt}{1.13M}} - 1\right) = v_1$$

now $a = \overline{W \sin \gamma} - \Delta C_d \cdot .001185 \cdot 1400 \cdot (169.3)^2 = W \sin \gamma - \Delta C_d \cdot 47700$

$b = -2 \cdot 169.3 \cdot .052 \cdot .001185 \cdot 1400 - \Delta C_d \cdot .001185 \cdot 1400 \cdot 2 \cdot 169.3 - .052 \cdot 1400 \cdot .001185 \cdot 169.3$

$b = -29.2 - \Delta C_d \cdot 562 - 14.63 = -43.83 - \Delta C_d \cdot 562$

$$\frac{-\overline{W\sin\gamma} + \Delta C_d \cdot 47700}{43.83 + \Delta C_d \cdot 562}\left[e^{-t(.042 + \Delta C_d \cdot .539)} - 1\right] = v_1$$

Calculate for run 2079

$\Delta C_d = + .007$ $\quad$ t = 27 sec elapsed = (30 sec on scale)

$C_d = .059$

$\overline{W\sin\gamma} = -1635$

$$\frac{1635 + .007 \cdot 47700}{43.83 + .007 \cdot .562}\left[e^{-27(.042 + .007 \cdot .539)} - 1\right]$$

$$\frac{(1635 + 334)}{47.77}(-.708) = -28.8 \text{ ft/sec} = -17.05 \text{ knots}$$

$K_1 = 82.95$ knots theoretically

$K_1 = 85.25$ measured

Calculate for Run 2080 nose down @ 22 Sec.

$C_{d_o} = .052$

$\overline{K_1} = 105$

$$\overline{C_L} = \frac{29792}{1400 \cdot .00256(1.152 \cdot 105)^2} = .566$$

$\overline{C_d} = .048; \quad \overline{\Delta C_d} = -.004$

$b = 1400 \cdot .001185 \cdot 169.3\ (-3 \cdot .052 + .008) = -41.6$

$$v_1 = \frac{1332 + .004 \cdot .001185 \cdot 1400 \cdot 169.3^2}{-41.6}\left[e^{\frac{-41.6 \cdot 22}{1.13 \cdot 923}} - 1\right]$$

$v_1 = -36.6\left[-.584\right] = 21.4 = 12.6K$

Ki = 112.6 knots theoretically

Ki = 109.69 measured

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APPENDIX NO. 1

PHOTOGRAPHS

Figure 1. PBY-6A NAVY 46683

Figure 2. 1 Kc Amplifier and 5-116 Oscillograph

Figure 3. Altimeter Bellows and Spring

Figure 4. Calibration Test of Altimeter

Figure 5. View of Starboard Pitot Tube

Figure 6. Velocity Plot of Airflow, PBY Nose

Figure 7. Oscillograph Records

Figure 8. Oscillograph Records

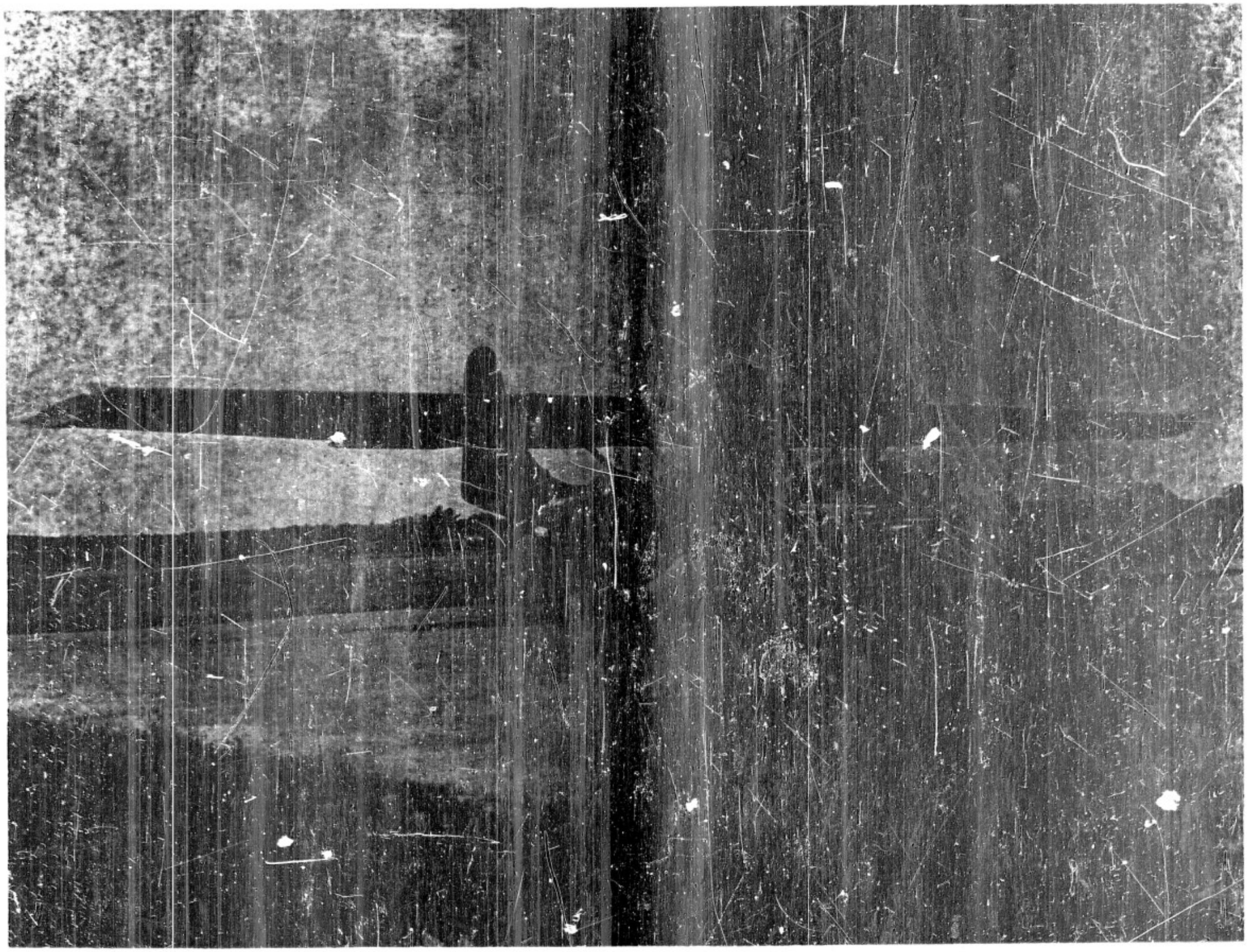

FIGURE 1. VIEW OF PBY-6A NAVY 46683

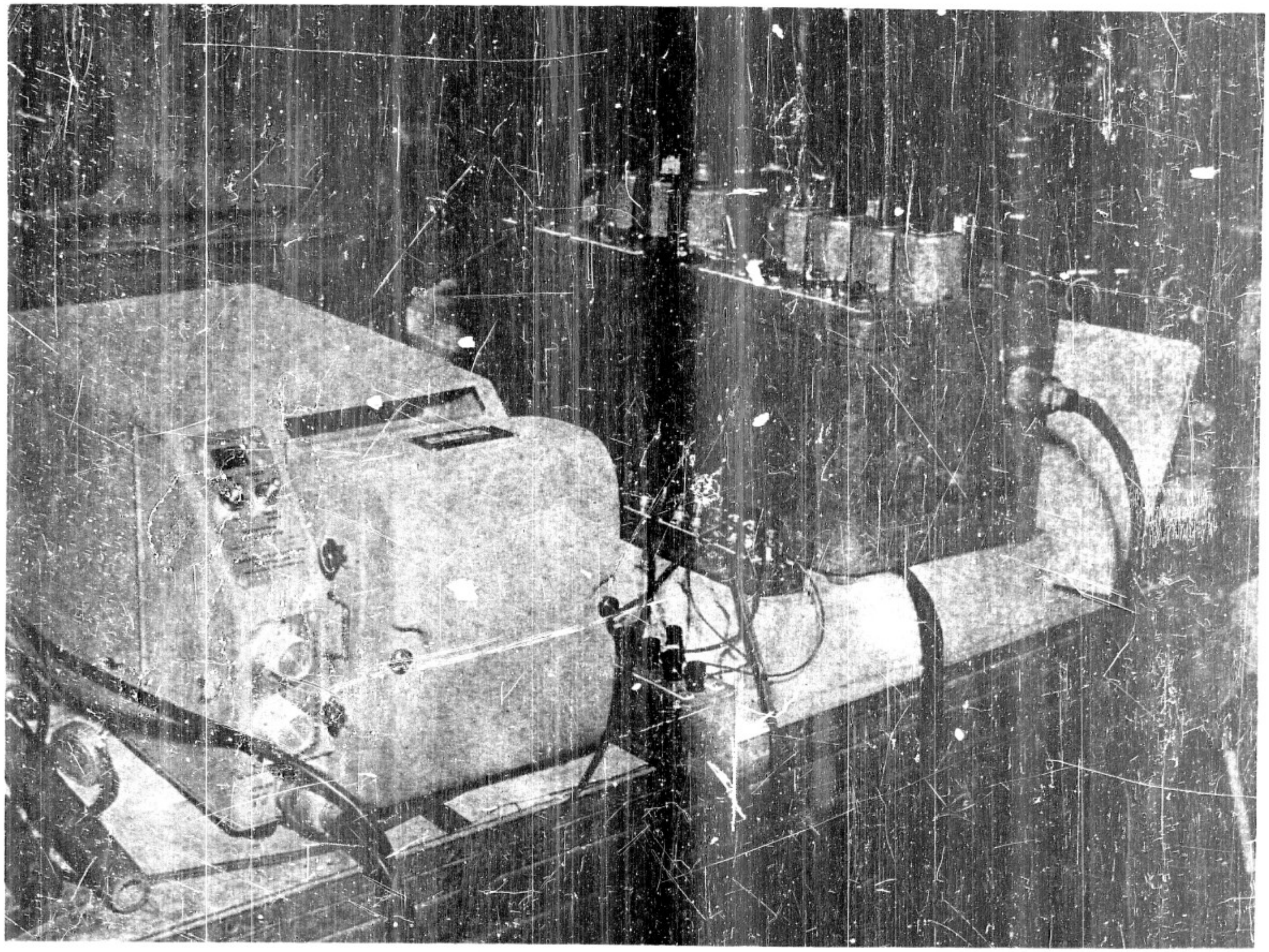

FIGURE 2. 1 KC AMPLIFIER AND 5-116 OSCILLOGRAPH

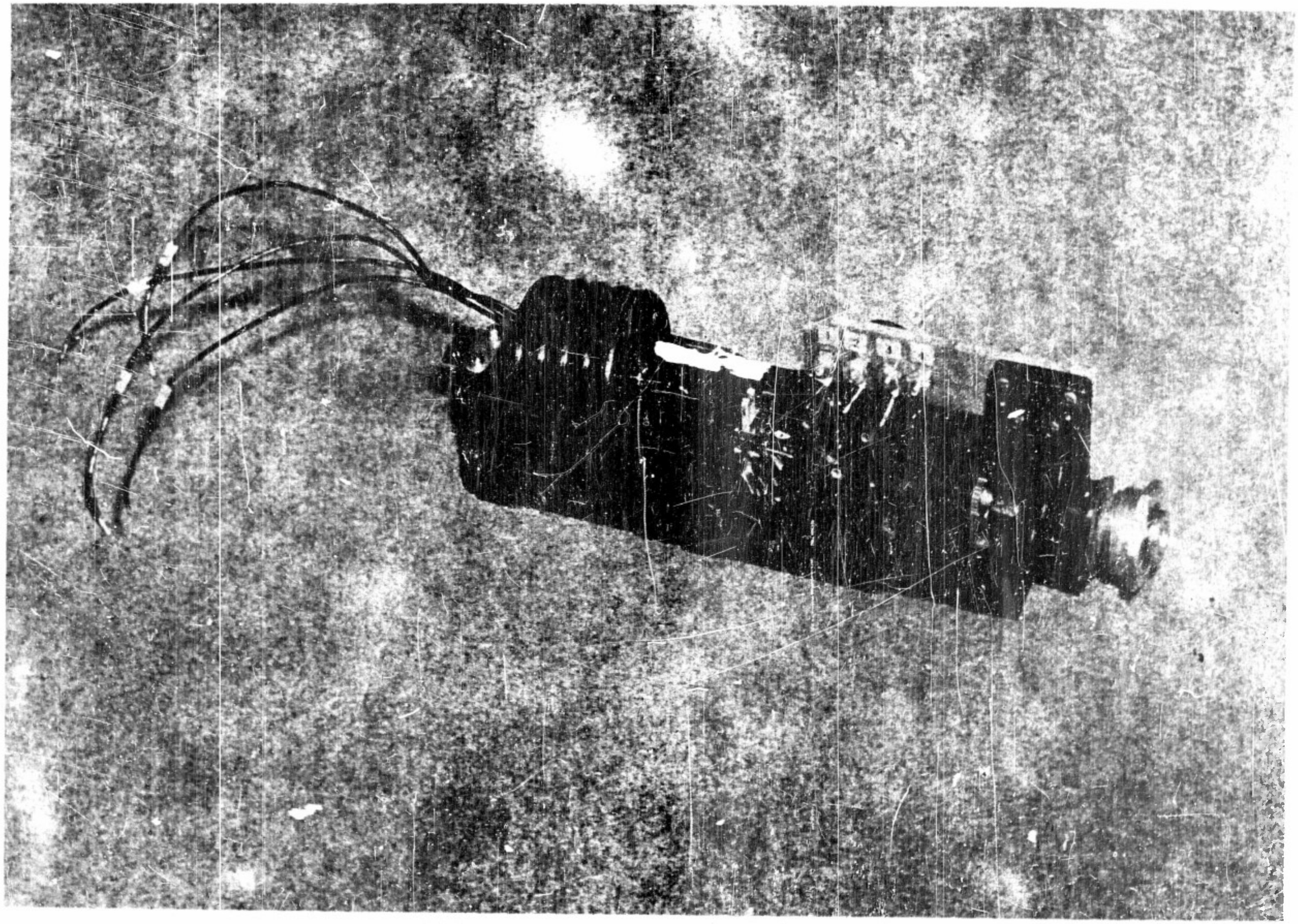

FIGURE 3. ALTIMETER BELLOWS AND SPRING

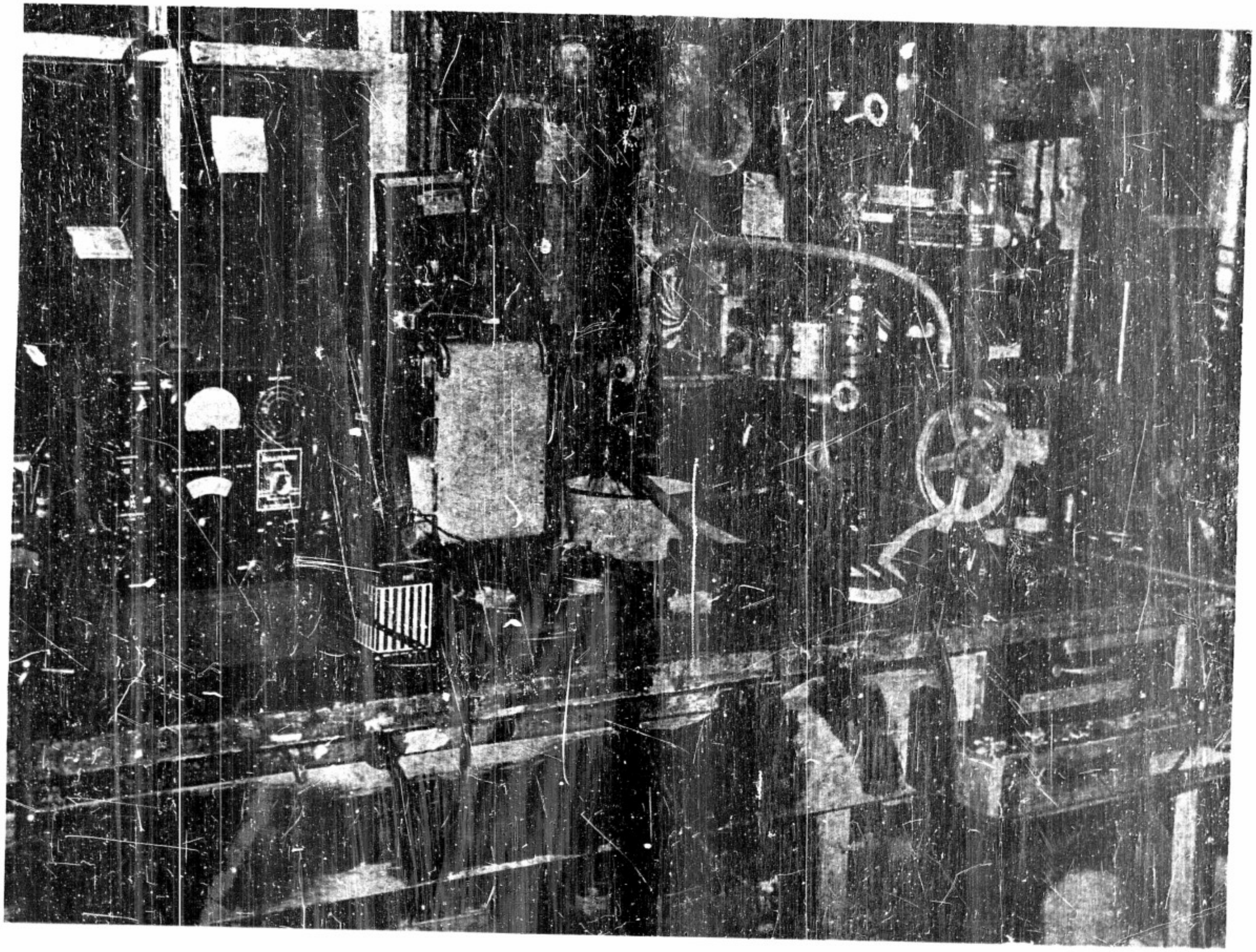

FIGURE 4. CALIBRATION TEST OF ALTIMETER

FIGURE 5. VIEW OF STARBOARD PITOT TUBE

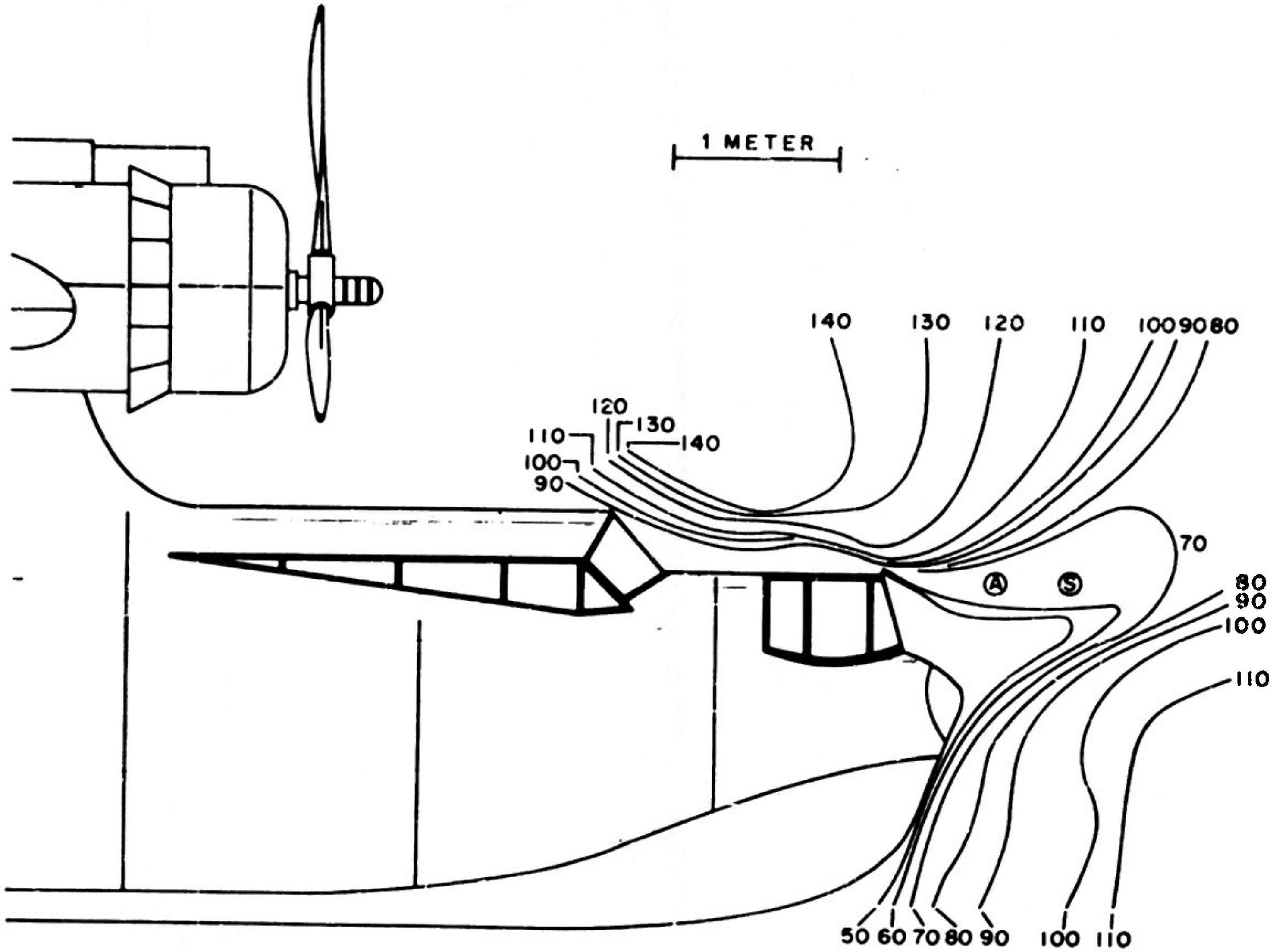

FIGURE 6. VELOCITY PLOT OF AIRFLOW, PBY NOSE

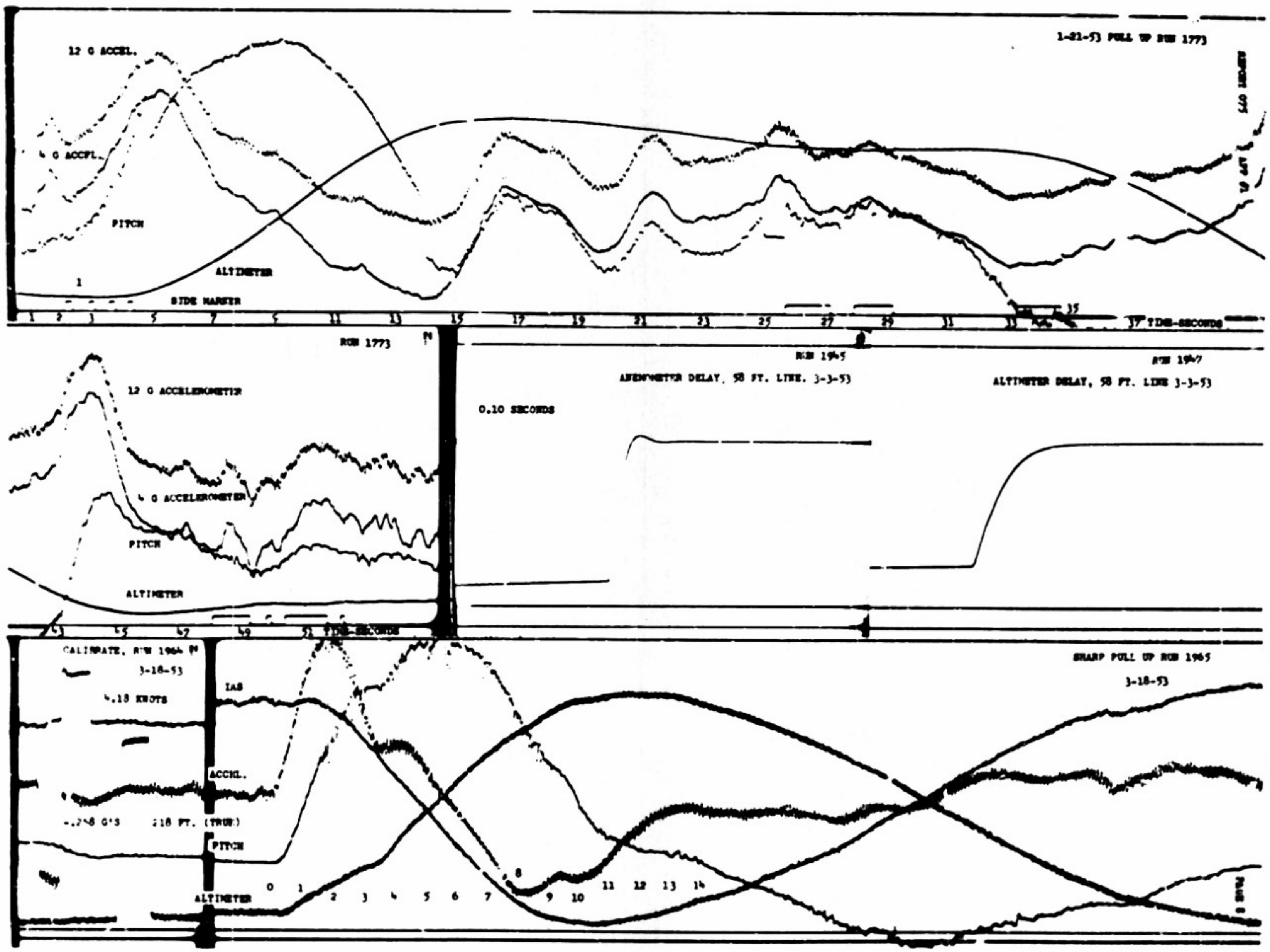

FIGURE 7. OSCILLOGRAPH RECORDS

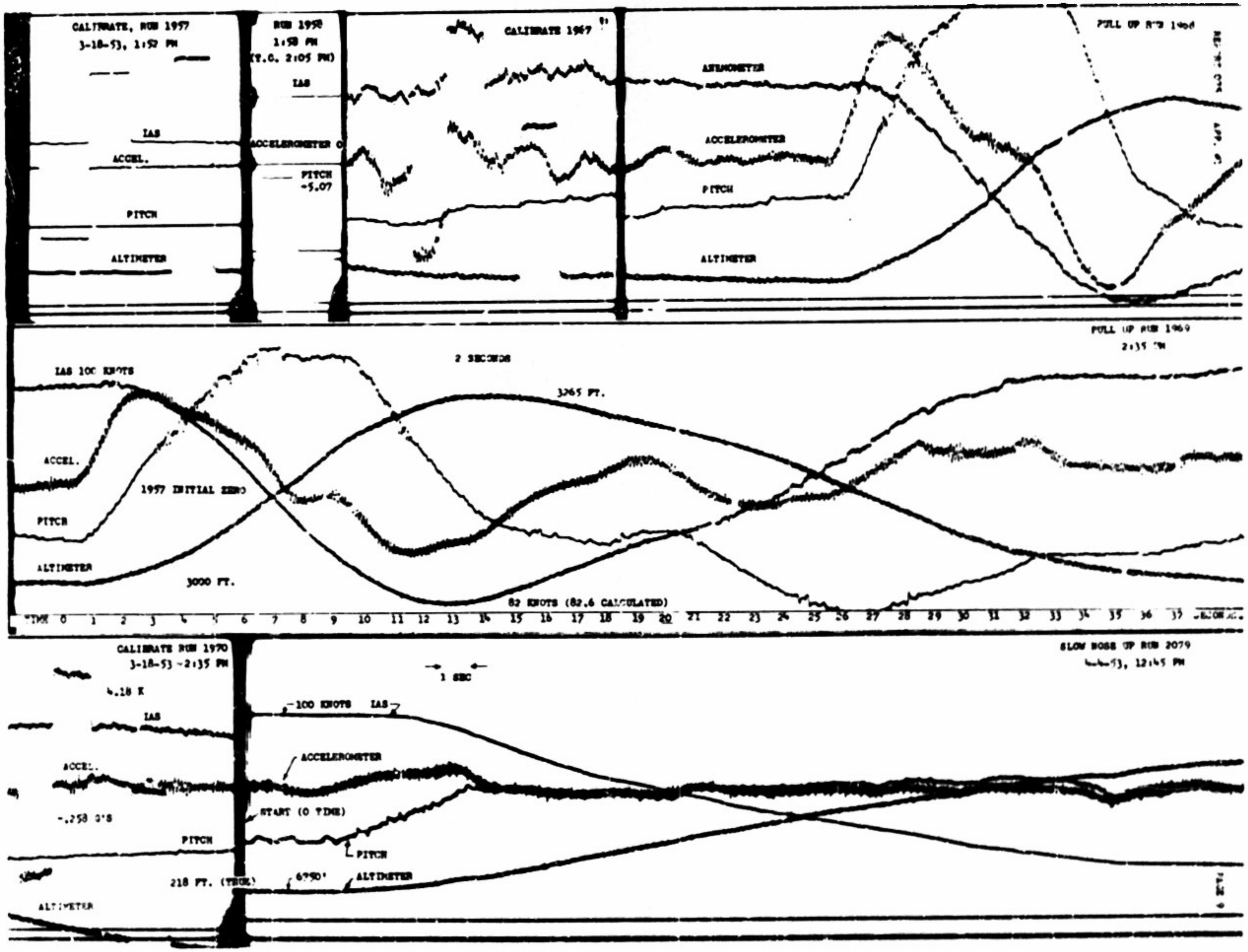

FIGURE 8. OSCILLOGRAPH RECORDS